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VIKING

BASELINE ORBITER CONCEPTUAL

DESIGN DESCRIPTION

Project Document No. 611-2

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(Addendum)

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PREFACE

The purpose of this addendum is to identify changes to the Viking Orbiter Baseline Design Document. This addendum delineates the authorized changes to the Orbiter baseline design and the latest understanding of the Orbiter Science payload.

Project Document No. 611-2 (Addendum)

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SECTION I

INTRODUCTION

The Orbiter defined in the "Baseline Design" document included a number of assumptions regarding the configuration of the Viking spacecraft. Since the issuance of the "Baseline" Document, an improved understanding of the Mission Objectives, and the Orbiter, Lander, and Launch Vehicle, capabilities has occurred. This improved understanding resulted in the Viking Project Office authorizing a weight reduction program so as to arrive at a Viking spacecraft weight that would provide a minimum launch period of 30 days. The resulting changes to the orbiter weight as well as other authorized changes consisting of a reduction in the total required ΔV from 1575 mps to 1420 mps, the incorporation of a one-hour solar occultation interval following the orbit insertion maneuver, and the use of "flight-loads" analysis for the structural design criteria, are described in Sections II and III of this addendum.

In addition to the above authorized changes the baseline model of the orbiter science instruments has also been updated. This updating provides a set of instruments for design integration studies that more nearly reflect the capabilities that will be required to meet the mission objectives.

The addendum format is similar to the format used in the Orbiter baseline design document. In the addendum each of the areas affected by design changes are referenced to corresponding sections of the baseline. A description of the design change implications is then presented.

SECTION II

VIKING ORBITER DESCRIPTIONS

A. ORBITER SEQUENCE

Section VII C. ORBITER SEQUENCE of the Baseline Document covering the orbit insertion sequence is modified to add up to one hour of sun occultation immediately following orbit insertion. Sun occultation capability at this point in the insertion sequence will be a requirement if low inclination orbits are to be used.

With power management and orbit insertion maneuver sequence modifications to the orbiter baseline sequence the orbiter design can support a one-hour solar occultation interval following the orbit insertion maneuver without serious impact on the orbiter design.

Key change in power management would include the turn-off of temperature control heaters during the orbit insertion maneuver. Changes to the orbit insertion sequence would consist of reducing the wait interval between turn commands and motor burn. The detailed modifications to the orbit insertion sequence have not, as yet, been finalized.

With the latest orbit insertion maneuver power profile (see appendix 1) the two baseline batteries can provide an off-sun capability of 3.5 hours. Allowing for one hour of solar occultation, 2.5 hours are available for performing the orbit insertion maneuver.

B. ORBITER WEIGHTS

The weight table presented here, appendix 2, supersedes Table 7D-1, Orbiter Subsystem Weight Summary contained in Section VII D. ORBITER WEIGHT of the Viking Orbiter Baseline Design Document.

Appendix 2 lists the Viking spacecraft weight breakdown as seen by the Orbiter. Major weight reduction has taken place in the propulsion subsystem. The reduced propulsion weight results from use of helium instead of nitrogen as a propulsion pressurant, an increase in the nozzle expansion ratio from 40:1 to 60:1, and a reduction of the total required ΔV capability to 1420 mps. Additional Orbiter weight reduction results from the use of light weight solar cells and thin cover glass, and the use of "flight-loads" analysis for structural design. Other Orbiter weight adjustments that have occurred are also explained in appendix 2.

SECTION III

VIKING ORBITER SUBSYSTEM DESCRIPTIONS

A. STRUCTURE

The structural design criteria presented here incorporates the philosophy of "flight-loads" analysis into the structural design philosophy of the Orbiter as contained in Section VIII A. STRUCTURE of the Orbiter Baseline Design Document.

The structural design criteria of the Viking Orbiter Baseline was based on an extension of the technology used for the Mariner series spacecraft. The Mariner structures were designed primarily to withstand loading conditions imposed by sustained sinusoidal preflight qualification tests. In order to effect a weight reduction, the Viking design will be based on a "flight-loads" analysis of the flight environment with a corresponding change to the preflight tests in order not to exceed the design conditions.

The structural elements proposed are consistent with a stress level not exceeding 30, 000 psi when subjected to a static loading of 8 g axial or a combination of 4 g axial and 2 g lateral. The "flight-loads" criteria do not yet exist but these static levels are estimated to be equivalent at this time.

B. POWER SUBSYSTEM

Changes to the baseline power subsystem Section VIII F. POWER SUBSYSTEM of the Viking Orbiter Baseline Design Document consist of the authorization to use light weight solar cells .008 inches thick and thin cover glasses .006 inches thick to reduce solar panel weight. With a total solar panel area of 115 ft^2 the light weight solar cells and cover glasses produce a 34 pound weight reduction when compared to the same panels covered with the standard Mariner 69/71 solar cells. Although the baseline is now changed, the selection of the solar panel design is still under review and therefore may be subject to further revisions.

C. PROPULSION SUBSYSTEM

Changes to Section VIII L. PROPULSION SUBSYSTEM of the Viking Orbiter Baseline Design Document consist of a change in the total ΔV requirement from 1575 mps to 1420 mps. Other design changes for reducing the propulsion subsystem weight both physically and by improving performance have also been selected and are discussed below.

For the latest spacecraft weight breakdown see appendix 2.

1. Substitution of Helium for Nitrogen as a Propellant Tank Pressurant

The pressurant has been changed from Nitrogen to Helium, effecting a decrease in the pressurant system weight of approximately 75 pounds.

Typically Nitrogen pressurant is preferred over Helium unless spacecraft mass is extremely critical, because of the lower leakage potential of Nitrogen systems. Both theory and experiments show that the ratio of leakage rate of Helium to Nitrogen is approximately three. In addition, Helium could cause leak path (hole) growth due to its higher molecular velocity. However, Gemini and Apollo propulsion subsystems utilize Helium pressurant as did the Surveyor vernier engines, proving that fabrication and component technology can be brought to bear to solve the potential leakage problems associated with Helium.

2. Increase Nozzle Expansion Ratio from 40:1 to 60:1

The RS-2101 engine for Viking will utilize a 60:1 nozzle expansion ratio, thus improving engine specific impulse by 4 seconds. This nozzle is 1.93 inches longer than a 40:1 nozzle.

The 40:1 nozzle currently available on the RS-2101 engine is fabricated of L-605. Studies have shown that a 60:1 nozzle can be fabricated of L-605 which will have acceptable thermal stress characteristics, but will have to be machined to a thickness of 0.010 inches to minimize vibration loads on the chamber/nozzle attach joint. The feasibility of this design is predicated on the vibration test criteria for the Viking engine remaining at a level equal to or less than the M'71 vibration test levels.

Anticipated problems, which are currently being assessed, are 1) fabrication feasibility of the light weight (0.010 inches vs 0.015 inches for M'71) nozzle, 2) structural integrity of the nozzle joint, and 3) susceptibility of the light weight nozzle cone to distortion due to vibration.

3. Use "Selected" RS-2101 Engine Injectors

During the TA and FA engine component testing program, performance tests will be conducted on approximately 14 injectors, as compared to a conventional qualification program which would include approximately 7* injectors. Injectors will be selected which meet or exceed the minimum specific impulse performance level of 289 seconds. A performance improvement of 2 seconds I_{sp} is provided by this selection process.

*Includes a total of four for TA qualification (1 $PTM_{subsystem}$, 1 $TA_{subsystem}$, 2 $TA_{component}$), plus three for flight deliveries.

D. DATA STORAGE SUBSYSTEM

The new storage handling requirements of the science instruments have resulted in necessary modifications to the data storage subsystem. The description contained in this addendum is a total rewrite of Section VIII O. DATA STORAGE SUBSYSTEM contained in the Orbiter baseline document.

The orbiter data storage subsystem consists of two identical digital tape recorders and their associated record, playback, and control electronics. Appendix 3 is a block diagram of the dual transport data storage subsystem. Each recorder has the capability of recording either the orbiter or the relayed lander data for increased reliability. Simultaneous recording of orbiter and lander data is possible by utilizing both recorders in a concurrent record mode. Effective orbiter data record rate will be on the order of 2.2 M bps. Record rates for lander data will be 16.2 K bps and 3.24 K bps. Playback rates will be 16.2, 8.1, 4.05, 2.025, and 1.0125 K bps. Orbiter data storage capacity of each recorder will be 5.0×10^8 bits. Lander data storage capacity of each recorder will be 2.0×10^7 bits for the 16.2 K bps record rate and 3.0×10^6 bits for the 3.24 K bps record rate. The tape recorder motor in each tape transport will be driven synchronously when recording orbiter data and in a servo-controlled mode at the lower bit rates. Jitter will be eliminated from playback data with the use of an output buffer.

DSS Parameters:

Weight	=	20 lbs/recorder
Power	=	22 watts/recorder
Storage Capacity	=	5.0×10^8 bits/recorder for orbiter data 2.0×10^7 bits/recorder for lander data (16.2 K bps record) 3.0×10^6 bits/recorder for lander data 3.24 K bps record)
Record Rates	=	2.2 M bps effective for orbiter data 16.2 K bps and 3.24 K bps for lander data
Playback Rates	=	16.2K bps, 8.1 K bps, 4.05 K bps, 2.025 K bps, and 1.0125 K bps

E. SCIENCE DATA SUBSYSTEM

The new data handling requirements of the science instruments have resulted in modifications to the Science Data Subsystem. The description contained in this addendum is a total rewrite of Section VIII P. SCIENCE DATA SUBSYSTEM contained in the Orbiter baseline document.

The Science Data Subsystem (SDS) is the data handling system for the science payload. It controls and sequences the science instruments, samples and converts the resulting data into digital form, accomplishes any necessary on-board processing, provides temporary (buffer) storage, and formats and routes the data to both the Flight Telemetry Subsystem (FTS) and Data Storage Subsystem (DSS) for direct or delayed transmission to Earth. The SDS exchanges commands with other subsystems aboard the spacecraft which pertain to the operation of the science payload.

The major differences between the Viking Orbiter SDS and the Mariner '71 DAS are due to the change in payload, increased flexibility required in sequencing the instruments and in constructing formats. This flexibility is required in order to sequence instruments in a manner which will make the best use of information obtained from previous spacecraft orbits and the Lander.

The SDS will produce three data streams. Two of these data streams will be provided to the Flight Telemetry Subsystem for direct transmission. The higher of the two (block coding mode) will be used when a 210-foot antenna is available and the low rate at other times. The high rate will be progressively lowered from 16.2 K bps to 8.1 K bps to 4 K bps as transmission conditions degrade during the progress of the mission. The low rate will be 133 1/3 bps.

The 16.2 K bps, 8.1 K bps, and 4 K bps data streams will consist of all science data, selected TV picture elements, and appropriate status and formatting data. The 133 1/3 bps Real Time (RT) stream will have selected portions of science, TV engineering status, and SDS status and formatting data. Flexible formats will be particularly valuable in making optimum use of these lower data rates. Appendix 4 shows a functional block diagram of the SDS.

The third stream of data consisting primarily of TV will be sent to the DSS on parallel lines. This digital TV data (an analog to digital conversion of the TV video signal) will be buffered and formatted by the SDS for recording by the DSS on parallel tracks. TV line format coding and other non-TV science data will be interleaved into each parallel line during the TV retrace time.

The science data must be buffered during the time that TV data are being recorded. A core memory similar to the Mariner '71 design will be used during these periods. The memory will contain a stored library of fixed formats as well as reprogrammable formats to accommodate different mission phases and contingencies. Coded commands received by the SDS to be used for controlling formats and sequencing will be routed to the buffer memory for storage and processing.

The third use of the memory will be in the implementation of a non-volatile frame count. This will guarantee continuity of science data throughout the 60 or more days of the orbital mission.

The volume required for the SDS will be approximately 500 cubic inches. The power will be about 20 watts, and the weight about 16 pounds. An operating temperature range of -20°C to +90°C will be defined as a design goal.

F. SCIENCE

The baseline model payload of orbiter science instruments has been updated to provide a set of instruments that more nearly reflect the capability required to meet the mission objectives. The new instrument requirements contained in this addendum replace the science instrument discussions contained in Section VIII R. SCIENCE of the Orbiter Baseline Document.

1. Imaging

The purposes of the orbiter imaging investigation are to aid in the selection of landing sites for the Viking landers and for future missions, to monitor the region surrounding the lander, and to study the dynamic characteristics of Mars. The geometric resolution of the orbiter imaging system is to be 30 meters per line or better at a reference altitude of 1000 km with image smearing from orbiter motion to be less than 50 percent of this resolution. Prior to lander separation, it is required from the near periapsis portion of the orbit to cover completely with contiguous pictures a swath at least 40 km cross-track by 500 km down-track on a single orbital pass. The near periapsis coverage requirement after lander separation is to obtain complete coverage with contiguous pictures of an area at least 50 km in radius centered on the lander on a single orbital pass. The capabilities provided to accomplish the above requirements will be utilized to accomplish the other cited purposes. To obtain both broad area and high resolution coverage, it is required that imagery be obtainable from the periapsis and apoapsis regions of the orbit using the same imaging system. The dynamic range is to be 100 to 1, and the sensitivity is to be sufficient to obtain pictures as close to the terminator as practical and to use three color filters.

The imaging subsystem consists of two identical telephoto cameras. Each camera is expected to be a cylinder approximately 38 inches long by 6 inches in diameter. Total instrument weight is expected to be 65 pounds. Primary power consumption is 45 watts average.

2. IR Radiometry

The objective of the IR radiometry investigation is to obtain surface temperature data to accomplish the same purposes as cited for the imaging investigation. The IR radiometer is to be boresighted with the imaging system, is to have the same coverage requirements, and is to be operable from the periapsis and apoapsis regions of the orbit. The temperature resolution is to be at least $\pm 1^{o}K$ at $200^{o}K$ and the measurement range is to be 150^{o}-$300^{o}K$. At a reference altitude of 1000 km, the spatial resolution is to be 10 km^2 or better after the lander has separated from the orbiter. Prior to separation, integration of the data over a larger area may be employed to achieve the required thermal resolution.

This instrument is expected to be approximately 8 inches square by 12 inches long. Total instrument weight is set at a maximum of 15 pounds.

3. IR Spectrometry

The objective of the IR spectrometry investigation is to determine the horizontal distribution of water vapor to accomplish the same purposes as cited for the imaging investigation. The IR spectrometer is to be boresighted with the imaging system, is to have the same coverage requirements, and is to be operable from the periapsis and apoapsis regions of the orbit. The water vapor measurement range is to be 5-100 μ precipitable water with a resolution of at least 1 μ at 20 μ. The spatial resolution is to be 20 km^2 or less after lander separation. Prior to lander separation, integration of the data over a larger area may be employed to achieve the required water vapor resolution.

This instrument is expected to be approximately 7 inches square by 25 inches long. Total instrument weight is set at a maximum of 20 pounds.

Appendix 1 - Revised Power Requirements - Orbit Insertion Maneuver (Battery Only)

	Orbit Insertion Maneuver	Orbit Insertion Maneuver Burn	Orbit Insertion Maneuver Unwind
Main Inverter (2.4 KHz)			
Flight Telemetry System	15.0	15.0	15.0
Flight Command	3.5	3.5	3.5
CC&S	22.5	22.5	22.5
Pyro	1.0	1.0	1.0
Power Distribution	2.25	2.25	2.25
Attitude Control System	51.2	51.2	51.2
Radio Frequency System	24.0	24.0	24.0
Gyro No. 1	10.0	10.0	10.0
SCN Electronics	4.7	4.7	4.7
Thermal Control No. 1	0	0	0
Data Storage System	0	0	0
Relay Radio Receiver	0	0	0
Relay Telemetry System	0.5	0.5	0.5
Antenna Pointing	3.0	3.0	3.0
Science	0	0	0
Sub Total	137.65	137.65	137.65
Efficiency	.898	.898	.898
Total	153.29	153.29	153.29
Gyro No. 2 (400 HZ)	9.0	9.0	9.0
Efficiency	.775	.775	.775
Total	11.6	11.6	11.6
SCN Motor	0	0	0
Reg. Converter (28 VDC)			
Valve	0	78.0	0
Gimbal	10.0	10.0	0
Efficiency	.87	.87	0
Total	11.4	101.15	0
Total Required Boost Regulator Output	176.29	266.0	164.89
Efficiency	.88	.886	.878
Total Boost Regulator Input	200.32	300.27	187.8
Unreg. Power Required			
Boost Regulator Input	200.32	300.27	187.8
Thermal Cont No. 3	0	0	0
Battery Charge	1.0	1.0	1.0
Boost Reg Fail Sensor	1.5	1.5	1.5

Appendix 1 - Cont.

Main Inverter (2.4 KHz)	Orbit Insertion Maneuver	Orbit Insertion Maneuver Burn	Orbit Insertion Maneuver Unwind
TWT	99.0	99.0	99.0
Relay Radio Transmission	0	0	0
Thermal Cont No. 2	0	0	0
Lander	15.0	15.0	15.0
Sub Total	316.8	416.77	304.3
Power Source & Logic Eff.	.97	.97	.97
Total Raw Power	327w	430w	314w

Appendix 2 WEIGHT SUMMARY (Table 7D-1)

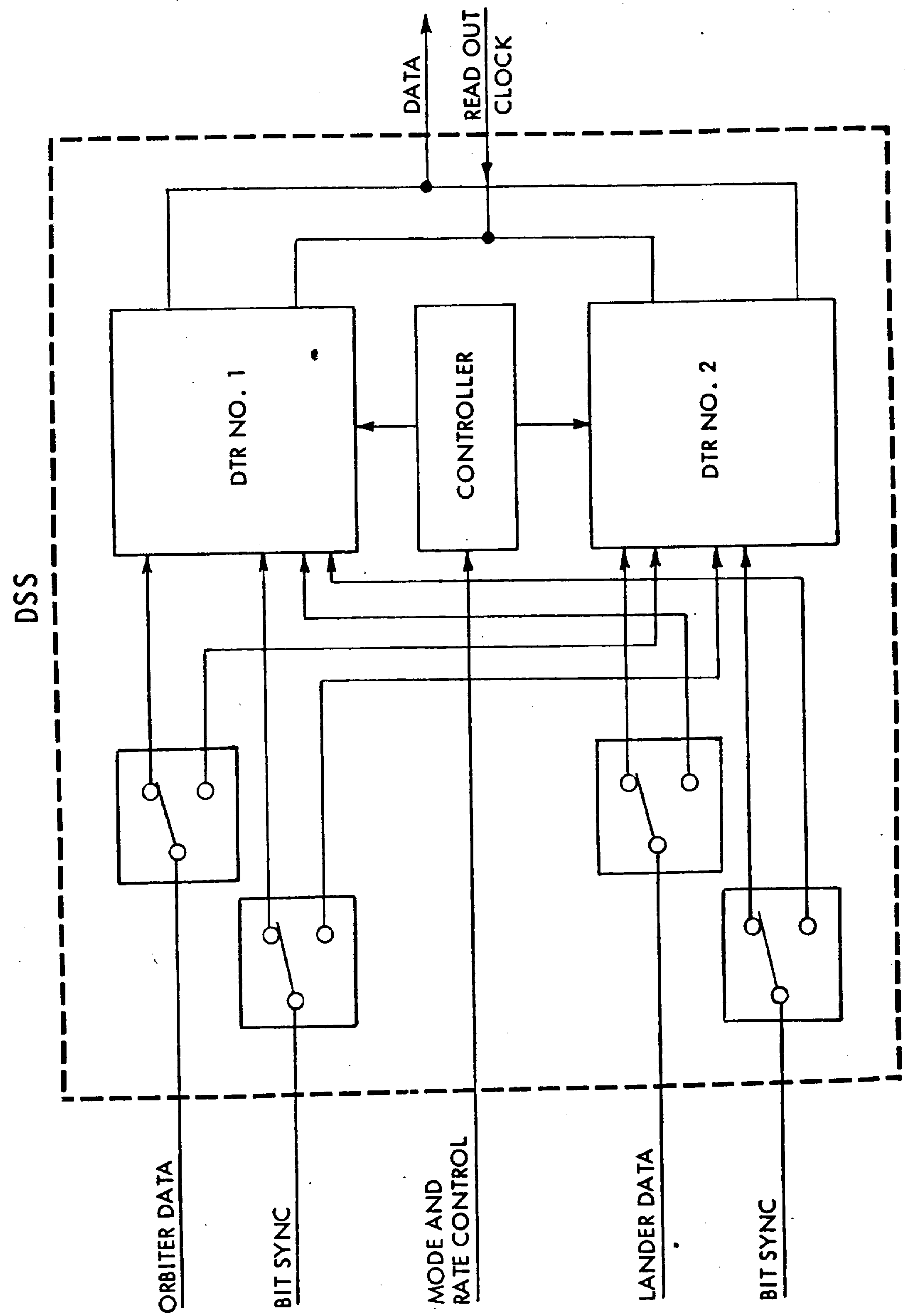
Subsystem	Basic Weight (lb)	Contingency Factor	ΔW from Baseline	Cause
(01) Structures	270.5	1.15	-14.0 lb	"flight-loads" analysis
(02) Radio	74.4	1.05	N.C.	
(03) Command	11.7	1.05	N.C.	
(04) Power	224.4	1.05	-34.0 lb	Light weight solar cells and cover glass
(05) CC&S	22.6	1.05	N.C.	
(06) Telemetry	26.0	1.10	N.C.	
(07) Attitude Control	101.9	1.10	-10.4 lb	Transfer of antenna control to the articulation control subsystem (15)
(08) Pyrotechnics	19.5	1.10	+4.5 lb	New estimate
(09) Cabling	109.0	1.10	N.C.	
(11) Temperature Control	32.0	1.15	+5.0 lb	Add propulsion thermostats
(12) Mechanical Devices	72.6	1.15	N.C.	
(16) Data Storage	40.0	1.10	N.C.	
(15) Articulation Control	21.9	1.05	+5.2 lb	Added antenna control & refined est.

Appendix 2 - Cont.

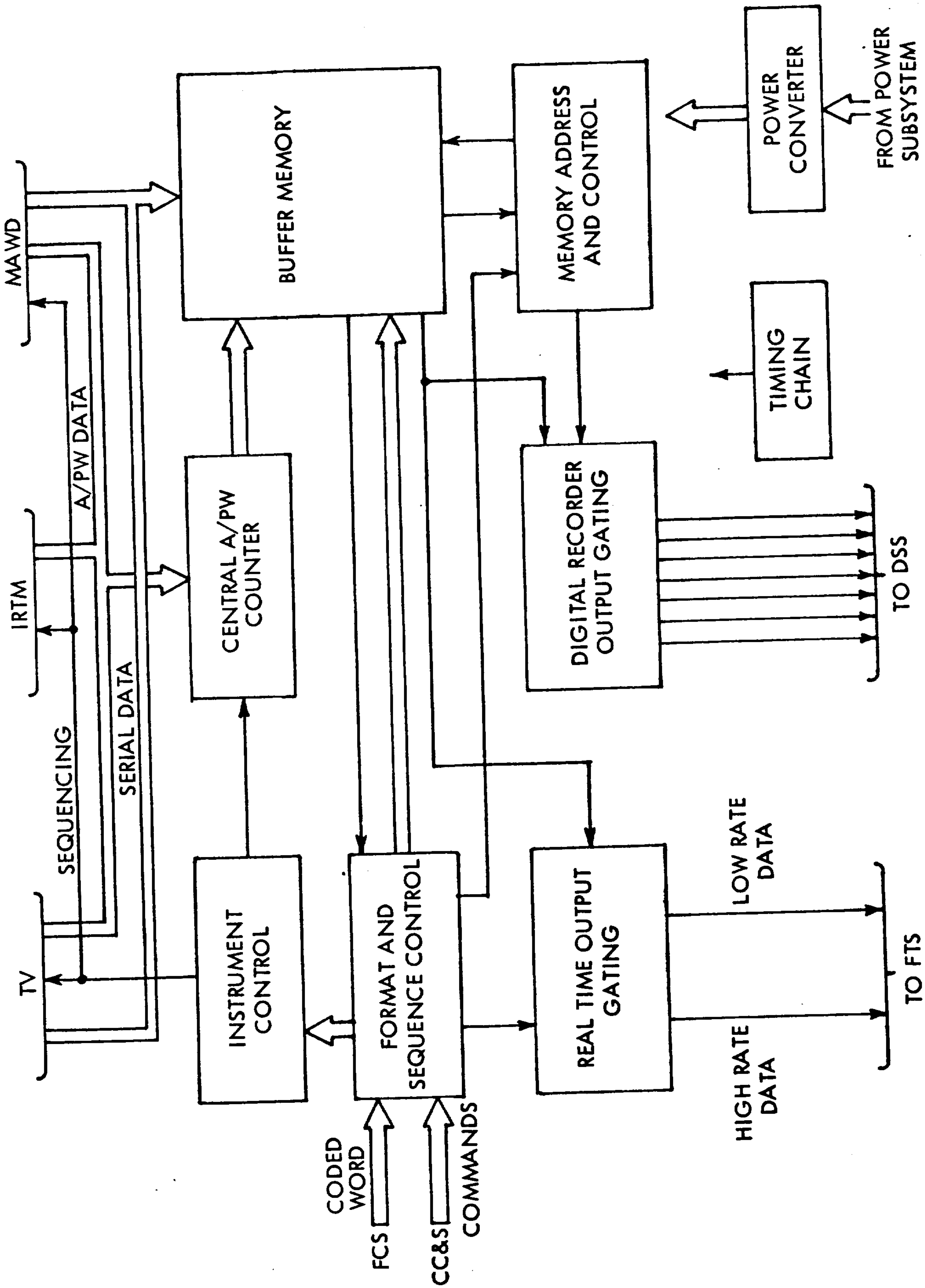
Subsystem	Basic Weight (lb)	Contingency Factor	ΔW from Baseline	Cause
(20) Science Data	16.0	1.15		
(36) Television	65.0	1.10		
(38) Infrared	15.0	1.10		
(39) Water Detector	20.0	1.10		
(52) Relay Radio	18.0	1.20	N. C.	
(56) Relay Telemetry	10.0	1.20	N. C.	

	Current Basic Weight (lb)		Contingency (lb)	Expected Weight (lb)
Viking Orbiter (less propulsion)	1170.5		164.5	1335.0
(10) Propulsion (inerts & residuals)	622.0		43.0	665.0
Usable Propellants	--		--	2784.0
O/L Adapter	40.0		6.0	46.0
Viking Lander Capsule	--		--	2194.0
Viking Spacecraft (Flight)	--		--	7024.0
Spacecraft-Launch Vehicle Adapter	129.5			
Destruct*	20.0	249.5	1.15	287.0
Stub Adapter & Accessory Equip.*	100.0			
Viking Spacecraft (Launch)				7311.0

*Shown for completeness only. Not JPL-supplied equipment.



Appendix 3. Data Storage Subsystem Block Diagram (Fig. 80-1)



Appendix 4. Science Data Subsystem Functional Block Diagram (Fig. 8P-1)

NOTICE TO PROPOSING CONTRACTORS,
REQUEST FOR PROPOSAL L10-9800

This document describes the Government furnished Viking Baseline Orbiter Conceptual Design. Included are the assumptions, guidelines and constraints that served as a focus for the design. The contractor shall use this document as required to support his effort during the proposal and pre-negotiation phases of this contemplated procurement. The final Orbiter design will be established by the Langley Research Center following contract go-ahead.

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Many personnel provided relevant information so that this report would be as comprehensive and as complete as possible. The names of the major contributors to this document are as follows:

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GLOSSARY

VIKING FLIGHT HARDWARE TERMINOLOGY

VLC	Viking Lander Capsule - All elements of the Lander Capsule (Bioshield, Aeroshell, Parachute, Lander, etc.) transported to Mars by the Orbiter.
VL	Viking Lander - Those hardware elements which accomplish the soft landing on the planet surface.
Bioshield Cap	The section of bioshield jettisoned prior to Lander separation.
Bioshield Base	The bioshield afterbody which remains attached to the Orbiter after the Lander separation.
VLC Adapter	The interstage structure which joins the VLC and VO, and which is integrally attached to and jettisoned with the bioshield base.
VO	Viking Orbiter - The flight article which transports the VLC and delivers it to the release point in Mars orbit.
VS/C	Viking Spacecraft (VS/C) - The VLC-VO combination excluding the LV to VS/C adapter.
Launch Vehicle and Spacecraft Adapters	(a) LV adapter - The adapter located below the field joint and integral with the Centaur. (b) S/C adapter - The adapter located above the field joint and which incorporates an isolation diaphragm.
LV	Launch Vehicle - The Titan/Centaur modified for the Viking mission.
Encapsulated Assembly	The Viking Spacecraft (VS/C), the S/C adapter, and the nose fairing.
Viking Space Vehicle	The entire vehicle that leaves the Earth's surface consisting of the Launch Vehicle and Encapsulated Assembly.

GLOSSARY

ABBREVIATIONS

The abbreviations used in this Viking document are as follows:

A	VLC adapter
A/C	Attitude control subsystem
A/D	Analog-to-digital
AFETR	Air Force Eastern Test Range
AGC	Automatic gain control
AJ	VLC adapter jettison event
AMB	Altitude marking radar
AOS	Acquisition of signal
APAC	Antenna pointing angle change
APS	Apsidal rotation angle
A/PW	Analog-to-pulse-width converter
BC	Bioshield cap
BCJ	Bioshield cap jettison event
B/H	Blockhouse
B/R	Boost regulator
Cabling	Cabling subsystem
CC	Coded command
CC&S	Orbiter central computer and sequencer subsystem
CPS	Capsule power system
CPT	Charged particle telescope
CRS	Capsule radio system
CS	Capsule separation event
DAS	Data automation subsystem

GLOSSARY

DC	Direct command
Devices	Mechanical devices subsystem
DFR	Dual frequency receiver
DMJ	Computer Instruction - Decrement minute and jump
DSIF	Deep Space Instrumentation Facility
DSJ	Computer Instruction - Decrement second and jump
DSN	Deep Space Network
DSS	Data storage subsystem
DTM	Development test model
DTR	Digital tape recorder
E + time	VLC entry plus a designated time
EMI	Electromagnetic interference
FCS	Flight command subsystem
F/F	Flip-flop
FSK	Frequency-shift-keyed
FTS	Flight telemetry subsystem
h_a	Altitude of apoapsis
HGA	High gain antenna
h_p	Altitude of periapsis
IB	VO insertion burn
INC	Orbit inclination
IRIS	Infrared interferometer spectrometer subsystem
IRR	Infrared radiometer subsystem
IRS	Infrared spectrometer
IRU	Inertial reference unit

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GLOSSARY

JPL	Jet Propulsion Laboratory
L + time	Launch plus a designated time
LCE	Launch complex equipment
LC&S	Lander computer and sequencer
LOS	Loss of signal
M + time	Maneuver initiation plus a designated time
M69	Mariner 1969 spacecraft
M71	Mariner 1971 spacecraft
MM71	Mariner Mars 1971 Project
MOS	Missions operations system
NAA	Narrow angle acquisition
NASA	National Aeronautics and Space Administration
NRZ	Non-return to zero
OD	Orbit determination
OI + time	Orbit insertion plus a designated time
OPS	Operations
P	Period of orbit
P_1 + time	First periapsis after orbit insertion plus a designated time
PAS	Pyro arming switch
PCM	Pulse-coded modulation
PIV	Planet in View
PLL	Phase-locked loop
PN	Pseudo-noise
Prop	Propulsion subsystem

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GLOSSARY

P_s + time	First periapsis following capsule separation plus a designated time
PSK	Phase-shift-keyed
PSL	Power source logic
PTM	Proof test model
Pwr	Power subsystem
Pyro	Pyrotechnic subsystem
QC	Quantitative command
RFS	Radio frequency subsystem
rms	Root mean square
RRS	Relay radio subsystem
RT	Real time
RTS	Relay telemetry subsystem
RZ	Return to zero
S + time	Launch vehicle/spacecraft separation plus a designated time
SAF	Spacecraft assembly facility
SAM	Structural analysis model
S/C	Spacecraft
SDS	Science Data Subsystem
SE	Support equipment
SIA	Solar illumination angle (sun elevation angle above horizon)
SIT	Separation-initiated timer
SPE	Static phase error
S/S	Subsystem
STC	System test complex

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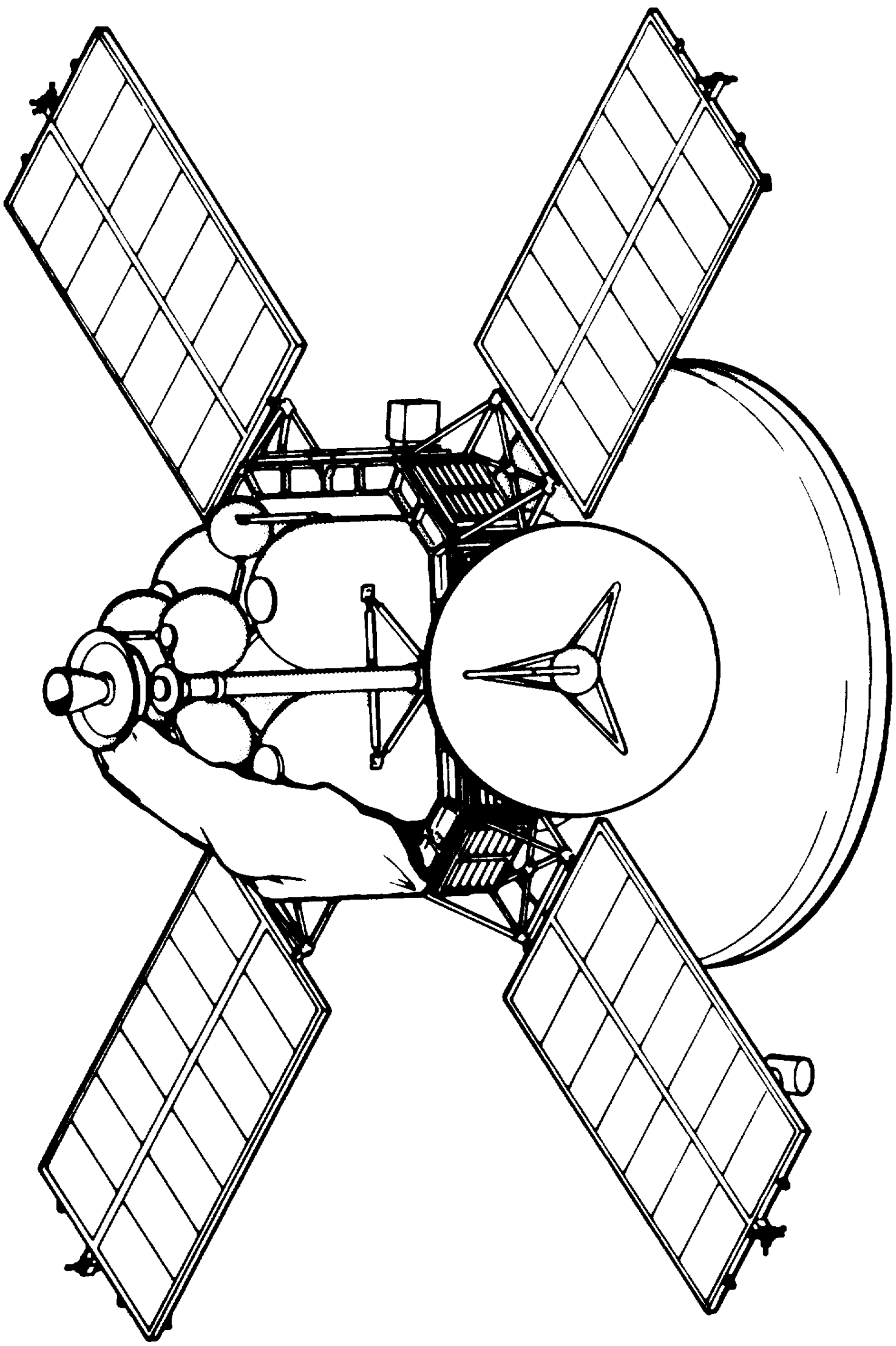
GLOSSARY

Structure	Structure subsystem
T - time	Countdown time prior to launch (clock stops during holds)
TA	True anomaly
TAB	Computer instruction - Transfer A to B
TB	Trim burn
TCM	Temperature control model
T/C	Temperature control subsystem
TD	VL touchdown
TLM	Telemetry
T/R	Transformer rectifier
TV	Television subsystem
TVC	Thrust vector control
TWT	Traveling wave tube
UVS	Ultraviolet spectrometer subsystem
V_∞	Hyperbolic excess velocity
VCO	Voltage-controlled oscillator
VL	Viking lander
VLC	Viking lander capsule
VO	Viking orbiter
VS/C	Viking spacecraft



Artist's Conception of Viking Spacecraft

SECTION I

INTRODUCTION

In response to a request by NASA's Langley Research Center, the Jet Propulsion Laboratory prepared this report which documents in detail an orbiter conceptual design which is to be used as a baseline design for the Viking mission. The Viking mission concept is to send two spacecraft to the planet Mars in 1973. These two spacecraft will consist of a Surveyor-type soft-lander mated to a Mariner 1971-class Mars orbiter. Hence, this document describes the conceptual design of such an orbiter.

The Viking mission is but one of a group of missions directed towards the exploration of the planet Mars using unmanned automated spacecraft. The first mission to explore the planet Mars was Mariner IV which was launched in November, 1964. This first mission obtained scientific information of the planet while performing a "fly-by" of the planet in July, 1965. Mariner IV obtained scientific information about the planet Mars which significantly advanced the knowledge of the planet beyond the measurements capable from the surface of the Earth. These scientific measurements included imagery of the Mars surface, an indication of no appreciable magnetic field or trapped radiation, and a measure of the atmosphere with the RF occultation experiment.

There are two additional missions in the implementation phase for the exploration of the planet Mars; Mariner '69 and Mariner '71. Mariner '69 is to be launched in early 1969 for an expected encounter in August, 1969. The Mariner '69 mission will send two spacecraft whose objectives are;

Primary -- to conduct flyby missions in order to make exploratory investigation of Mars which will set the basis for future experiments, particularly those relevant to the search for extraterrestial life.

Secondary -- to develop technology needed for the succeeding Mars missions.

The experiments chosen for the '69 mission were a dual resolution imagery, IR radiometry, IR spectrometry, UV spectrometry, RF occultation and celestial mechanics.

The Mariner '71 mission will place two spacecraft into Mars orbit. The '71 spacecraft are to be launched in May 1971 with an expected encounter in November, 1971. The mission objectives of Mariner '71 are to provide broad topographic and thermal coverage of the planet, to study the seasonal variations in the atmosphere and on the surface, and to perform other long term dynamic observations. Mariner '71 design will permit the study of dynamic characteristics of the planet from orbit for a minimum period of 90 days. During this period, scientific data will be obtained on the atmospheric composition, density, pressure, and temperature. Data will also be obtained on the surface composition, temperature, and topography. Approximately 70% of the planet's surface will be covered by some of these scientific measurements.

The science experiments chosen for the Mariner '71 mission were; dual resolution television, ultraviolet spectrometer, infrared interferometer spectrometer, infrared radiometer, charged particle telescope, dual frequency receiver, X-ray particle detector, and celestial mechanics.

The Viking mission program management is under the direction of NASA's Office of Space Science Applications, Planetary Programs Division. The Langley Research Center has the over-all project management responsibility and direct responsibility for managing the lander portion of the project. The Jet Propulsion Laboratory has the management responsibility for the orbiter portion of the spacecraft.

A general mission summary of the Viking mission used for baseline orbiter conceptual design is presented in the following section of this document, and the remainder of the document treats almost exclusively the orbiter baseline concept with a section in the appendix dealing with the orbiter-lander interface area.

SECTION II

MISSION SUMMARY

The Viking mission* concept used for the baseline conceptual design would be to launch two spacecraft about 10 days apart, using the Titan/Centaur as the launch vehicle in mid-1973. Each of these two spacecraft would consist of a Surveyor-type lander and a Mariner-class orbiter. These spacecraft would arrive at the planet Mars, still separated by about 10 days, approximately seven months after launch. During the interplanetary cruise period the orbiter would supply power and communications support to the lander.

Upon arrival at the planet Mars, the orbiter propulsion system would be used to place both the orbiter and lander into orbit about Mars. During the initial orbital period, the orbiter would provide reconnaissance of potential landing sites. Subsequent to the landing reconnaissance phase, but prior to the release of the lander from the orbiter, the orbit would be trimmed to become synchronous with the selected landing site. After a suitable landing site has been selected and orbit trimmed, the lander would be detached and would soft-land using the terminal landing phase techniques developed for the Surveyor and the Apollo Lunar Module. The orbiter would perform as a relay satellite during the entry phase of the lander into the Martian atmosphere to gather scientific information of the entry from the lander.

After the lander has reached the surface of the planet the orbiter will continue to provide relay support for the lander. This support will continue for a period of at least three days. At the conclusion of these three days of functioning as a Mars relay satellite, the orbiter would be free to function independent of the lander and perform further reconnaissance of the Mars surface for the next 90 days, except for reestablishment of the relay link at least three times in the 90-day period.

The lander, once upon the surface of Mars, would proceed to carry out the assigned lander scientific objectives and transmit the data via relay link or directly to Earth. The lander would continue to transmit scientific information to Earth for a period of at least 90 days.

*For actual Viking Mission definition refer to Viking Project Document M 73-101-3

SECTION III

ORBITER OBJECTIVES AND CONSTRAINTS

The primary objective of the Viking mission is to obtain information pertaining to the existence and nature of life on Mars, the atmospheric and surface characteristics of the planet, and the nature of the planetary environment. The specific objective of the orbiter portion of the orbiter-lander spacecraft are focused upon delivery of the lander into orbit around the planet, to provide orbital reconnaissance for the determination of suitable landing sites, and to support the lander as a relay satellite. In addition, the orbiter shall perform orbital reconnaissance of Mars to characterize the planetary environment.

The specific guidelines and constraints which were imposed on the Viking mission, as affecting this baseline conceptual design, were:

1) Two Titan/Centaurs as launch vehicles.
2) Single launch complex (serial launch)
3) Two spacecraft -- one lander and one orbiter per spacecraft.
4) Out-of-orbit delivery of the lander.
5) Nominal lifetime of lander and orbiter; 90 days after lander touchdown.
6) Orbiter acquires lander and orbiter data with lander data as primary.
7) Orbiter design to incorporate relay satellite capability to support lander during entry phase, initial landing phase, plus three selected periods during mission lifetime.
8) Lander has direct-link capability.
9) Three 210' Deep Space Net antennas are available.
10) Orbiter conceptual design compatible with orbits specified by Langley Research Center:
    a) Period - 24.6 hours
    b) Periapsis - 1,000 km
    c) Apoapsis - 33,000 km
    d) Inclination - 40°

11) Orbiter conceptual design must be compatible with possible Sun occultations.

12) Maximum lander weight of 1800 lbs.

13) Orbiter impulsive ΔV capability of 1350 m/sec for orbit insertion maneuver.

14) Orbiter capable of several orbit trims.

15) Orbiter Science Payload - assumed for this conceptual design

   a) TV

   b) IR Radiometer

   c) IR Spectrometer

   d) IR Multidetector Spectrometer

16) Orbiter relay link capable of 10^7 bits/day during relay periods.

17) Orbiter capable of ranging.

18) Orbiter telecommunications capable of supporting engineering telemetry during all maneuvers.

SECTION IV
ORBITER SCIENCE OBJECTIVES

NOTE:

The orbiter science objectives section of this document prepared by the Jet Propulsion Laboratory in the development of the orbiter conceptual design has been purposely removed at the request of the Langley Research Center. The proposers on L10-9800 are requested to refer to the Viking Project Mission Definition document number M73-101-3, page 6, section 4.2, for orbital science objectives.

SECTION IV

ORBITER SCIENCE OBJECTIVES

A. OVERALL MISSION OBJECTIVES

The scientific goals of the NASA planetary exploration program with specific relation to Mars are to acquire information relevant to the origin and evolution of the planet and to the origin, evolution, and nature of life, and to apply this information to the understanding of our own home planet and of life upon it. The objectives of the Viking lander and orbiter, in turn, are to contribute to these program goals in ways that are appropriate to each.

Because of the acknowledged overriding importance of the search for life on Mars and the urgency of making major progress on this search before it is compromised or confused by external contamination of the planet, it is proper for this first Viking mission that all objectives should be evaluated in terms of their contribution to this search. By this it is not implied that our interests are entirely limited to this objective, since almost all aspects of Mars are relevant to the biological question. On the contrary, the primacy of the biological objective will place no great limitation on the type and use of the information to be gained.

The principal biological questions to be asked about Mars in this and subsequent missions are:

1) Are there any visible signs of life?
2) Does the chemistry of the surface and atmosphere suggest the presence of life?
3) Is it possible to establish the presence of active biochemical processes?
4) What are the main environmental constraints for life on Mars?

B. GENERAL ORBITER OBJECTIVES

The Viking orbiter's mission will comprise three principal tasks:

1) To determine the optimum landing site
2) To support and supplement the scientific observations of the lander
3) To conduct additional independent scientific investigations

Although the tasks will in general be performed in the order listed, they will be overlapped and interwoven to an extent.

It is generally acknowledged that information obtained from Mars orbiters is not likely to give conclusive evidence either for or against the existence of life on the surface. Hence, the prime scientific objective of the Viking orbiter is to place the lander in the most advantageous position for it to obtain the evidence. Landing site surveillance is discussed in more detail in section V. The site should have two main characteristics: it should be safe for landing, and it should be of potential biological interest. The surveillance should seek to maximize the probability that these criteria are satisfied. The properties that are commonly assumed to identify favorable sites for biological exploration and that can be detected from an orbiter are:

1) Locally elevated surface temperature
2) Unusually high water concentrations
3) Visual changes, particularly of a seasonal nature, in surface features such as color or albedo

In support of the lander's scientific objectives, the orbiter will relay the lander's data output back to Earth. In addition, by means of its own scientific instrumentation, it will:

1) Attempt to determine precisely the location of the lander in relation to visible topographic features
2) Define the environment of the lander, observing and measuring as many environmental parameters as possible

3) Monitor temporal changes in this environment so that specific lander experiments can be programmed appropriately to obtain maximum information.

The scientific mission of the orbiter will be chosen from among the objectives discussed in the following section, constrained by the capabilities of the instruments that are carried, and guided by knowledge from other sources. In the listing of scientific objectives for Viking, no account has been taken of the fact that certain of these may already have been fulfilled to a degree by earlier Mars missions. Few if any fundamental questions about the planet are likely to be completely answered by the early mission, and even regions that have been very thoroughly examined will require reobservation because of the great importance of temporal variations as indicators of biologically interesting areas.

In the following discussions of specific scientific objectives for the orbiter, an attempt has been made to indicate the relative importance when it is clear, but practical considerations will determine which goals can actually be pursued on this mission. Remarks relative to possible instrumentation are included, but no prejudgment of their scientific or engineering feasibility is implied.

Relevance of the objectives to the pre-landing (P), support (S), or independent (I) phases of the orbiter's mission are indicated by code letters. In many cases, of course, whether a particular investigation "supports" the lander will depend upon where or when the observations are made or upon what phenomena are actually discovered.

C. SPECIFIC OBJECTIVES -- CONFIGURATION AND ACTIVITY

Objectives related to structure and processes occurring below the surface of the planet include the following.

1. Internal Mass Distribution and Dynamical Flattening (I)

If it is feasible to leave the orbiter free of nongravitational forces (e.g., intentional orbit corrections) for a sufficiently long tims, the doppler and range tracking data may be able to map the gravitational field around Mars more accurately than and in different regions from what has been done by observation of the Martian satellites. This would determine the gravitational flattening, and might uncover mass concentrations similar to those recently detected inside the Moon.

2. Geometrical Asymmetry (I)

The oblateness of the mean surface is controversial because it can be measured only inaccurately from Earth. The question has an important bearing on the long term mechanical strength of the interior. Visual photography and radar altimeter measurements could provide such information.

3. Differentiation (I)

The degree of chemical differentiation of the planet is one of its most fundamental properties and crucial to the understanding of its formation. We should particularly like to know whether Mars has internal density discontinuities and a liquid core. If near-spherical symmetry exists, the orbiter tracking data probably cannot detect differentiation. Measurement of an intrinsic magnetic field, if it exists, would give some clues, however. A measurement of the quantity of radioactive isotopes (potassium, thorium, and uranium) in the surface would also give an indication of the degree of near-surface, crustal differentiation. A gamma-ray spectrometer could provide this measurement.

4. Orogenic and Volcanic Activity (P, I)

The search for large scale orogenic and volcanic activity is obviously a prime objective of the optical imaging experiments. Extensive mountain ranges, plateaus, basins, faults, or graben-like features would indicate that the planet is orogenically active. Active fissures and thermal springfields could be indicative of smaller scale processes occurring within the crust.

Evidence for volcanic extrusion or active emplacement of intrusive igneous bodies might also be provided by thermal mapping with an infrared radiometer or by a spectroscopic identification of anomalously high local concentrations of a characteristic effusing gaseous component or of water. The biological implications of vulcanism are profound, since it can provide both water and warmth.

5. Magnetic Field (I)

If an intrinsic magnetic field is detectable, determination of the magnitude, direction, eccentricity, and multipole nature of the magnetic moment will provide revealing clues to the internal structure.

D. SPECIFIC OBJECTIVES -- SURFACE

Mapping of the surface (P, S, I), both extensively and intensively, is the prime objective of all early orbiters. In order that the maps contain the maximum possible amount of information, a diversity of instruments is required, and every effort should be made to achieve spatial and temporal correlation among them. Visual mapping, preferably multicolor, locates the basic features. Stereoscopic techniques and shadow observations can give altitude information. Radar mapping can provide more details on altitude, and regions of unusually low altitude (where the atmospheric pressure and temperature will be higher) have more than average biological interest. An IR spectrometer with high spatial resolution could also give altitude based on the range in CO_2.

Other surface properties that should be included in the mapping are:

1) Temperature, as indicated by infrared emission (P, S, I). Thermal mapping might reveal vulcanism or hot springs. It will also provide information about the possible physical states of water. Day and night maps should be correlated if possible.

2) The presence of surface water, subsurface water and permafrost (P, S, I). It seems not improbable that water may betray its presence to the TV camera as frost deposits or stationary clouds. In addition, its detection on and in the ground is feasible from an orbiter using radar operating at or below 10^7 Hz.

3) Gross rock composition (I). Spectrometric scanning in the visible and near and mid infrared can yield this information. Electronic absorption bands in the near infrared caused by the transition metals (iron in particular) vary in position according to the volume state of the element. Iron-bearing silicate minerals can be identified by this method. Frequencies of oxygen-silicon lattice resonances in the 8-14 micron region dependent on minerology and crystal structure. Materials such as basalt, chondrite, granite, etc., can be distinguished by these methods.

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4) Spectral reflectivity and polarization properties (I). Temporal changes in all these surface characteristics (both diurnal and long term) are of special importance. Hence, a premium should be placed on observation of the same areas at the beginning and end of the orbiter lifetime, and a reobservation of same areas observed by the '71 orbiter.

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E. SPECIAL OBJECTIVES -- ATMOSPHERE

1. Chemical Composition (S, I)

Atmospheric components in the upper atmosphere can be detected by UV spectroscopy, which generally detects emission lines from atoms and absorption lines from diatomic molecules, including ions and free radicals. Altitude profiles can be obtained at high altitudes. Lower down this technique is supplemented by IR spectroscopy, which detects absorption lines from polyatomic molecules.

2. Water Vapor (P, S, I)

Water is considered to be probably the key to the biological question, and an objective of surpassing importance is to obtain its four-dimensional distribution (i.e., in latitude, longitude, altitude, and time) in the lower atmosphere. Water vapor is detectable by an IR spectrometer or a filtered IR radiometer and perhaps by a microwave radiometer. Visual detection of local clouds or snow associated with igneous activity is also a possibility.

3. Trace Constituents and Nonequilibria (S, I)

Since living organisms exchange materials with the atmosphere, it is believed that the atmosphere will contain evidence of this interaction. This idea is reflected in two theories: that there will be a shift in the thermodynamic equilibrium of the atmosphere, and that there will be a diurnal cycling of some minor constituents. Both these effects may be detectable spectroscopically.

4. Meteorology (P, S, I)

Measurement of the gross meteorological properties such as temperature profile, circulation, and precipitation is an important objective. Orbiter observations will complement lander data. Specifically, IR spectrometer techniques will yield temperature profiles while the visible imaging may reveal the presence and motion of clouds or precipitation. Simultaneous observation of a cloud from the orbiter and the lander is an exciting prospect. Orbiter observations of cloud or storm motion might be used to program the schedule of lander experiments.

5. Atmospheric Aerosols (S, I)

The presence of airborne aerosols (suspensoids consistency of condensates of minor gaseous constituents and particulate matter) could severely affect the interpretation of IR and imaging data. A search for and identification of aerosols in the atmosphere must be included in the mission objectives.

6. Pressure and Temperature Profiles (I)

We should like to know how the altitude profiles of atmospheric pressure and temperature vary with latitude, time of day, and season. UV spectroscopy of Rayleigh-scattered radiation in the twilight layer gives the density profile there; hence UV spectroscopy can determine the latitude and seasonal effect at twilight. High resolution IR spectroscopy, in principle, gives density and temperature profiles along the scan path in daylight if the resolution is high enough and there are no significant concentrations of other scatterens in the path. Microwave radiometry may also be a possibility. Density profiles can be deduced from the effect of the atmosphere on the telemetry signal when the spacecraft is occulted by the planet or on solar photons (X-ray, UV, or visible) when the Sun is occulted.

7. Photochemical Processes (I)

The photochemical reaction products in the upper atmosphere are detectable by a UV spectrometer. They may aid in the interpretation of composition analysis or be indicative of the surface environment.

8. Exosphere (I)

A question of importance to understanding the history of the atmosphere, is that of the mechanism by which light atoms escape from the atmosphere and the rate at which they do so. On Earth the escape rate is controlled by the exosphere temperature, which can be measured by radiometers or spectrometers in the UV. On Venus it appears that exospheric heating is largely the result of impact of the solar wind on the upper atmosphere, and the same is probably true of Mars. If so, orbiter measurements with a magnetometer and a plasma probe would indicate the fact.

F. ADDITIONAL OBJECTIVES

Three categories of additional objectives can be identified. Each is intrinsically interesting and could be investigated by an orbiter such as Viking. Their relationship to the central scientific mission of Viking appears somewhat less direct, however, than those of the three categories discussed above.

1. Ionosphere (I)

The altitude profile of the electron density in the ionosphere can be investigated by its effect on radio waves, either by the technique of the "topside sounder", as has been done on Earth orbiters, or by the dual-frequency occultation technique utilized on Mariner Venus and Mars missions. If the solar wind actually strikes the top of the atmosphere, these measurements should reveal it.

2. Environment (I)

Various environmental factors can be named. The effects of solar radiations, including ultraviolet, X-rays, solar protons, etc., can either be calculated or investigated on other space missions with more than sufficient accuracy to define their effect on Mars. The same can be said of cosmic rays. The solar-wind interaction has already been mentioned; it could and should be investigated on some Mars orbiter -- not necessarily this one. The micrometeoroid flux near Mars is unknown but probably not important.

3. Satellites (I)

The two Martian moons are clearly interesting and important subjects for future investigation, but they appear to have no relation to this Viking mission, barring the unlikely chance that we might get a picture of one, or the even more unlikely chance of a collision.

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SECTION V
MISSION ANALYSIS

NOTE:

The mission analysis section of this document prepared by the Jet Propulsion Laboratory in the development of the orbiter conceptual design has been purposely removed at the request of the Langley Research Center. The proposers on L10-9800 are required to develop and supply such analysis with respect to the mission objectives specified in request for proposal L10-9800. All pertinent guidelines and constraints have been included elsewhere in the request for proposal.

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SECTION V

MISSION ANALYSIS

A. INTRODUCTION

This section is devoted to the mission analysis considerations that influence the spacecraft design. The broad mission guidelines are given in sections III and IX. However, many additional guidelines and assumptions are made in this section to allow the spacecraft design to proceed.

Included in this section are interplanetary trajectory selection and characteristics, orbit selection and characteristics, landing site surveillance and characteristics, orbiter propulsion velocity requirements and constraints, and launch vehicle system.

B. INTERPLANETARY TRAJECTORY SELECTION AND CHARACTERISTICS

1. Content

This section defines the Earth-to-Mars trajectory space which the spacecraft design must accommodate. The required areocentric trajectory space is defined in section V.C.

2. Trajectory Characteristics

The trajectory of the spacecraft may be divided into three periods: near-Earth, heliocentric, near-planet.

The characteristics of the near-Earth trajectory are determined by the launch date, arrival date, and launch azimuth. Specification of launch date and arrival date defines the required energy and asymptote declination of the departure hyperbola.

The spacecraft trajectory characteristics during the heliocentric period are defined by the launch date and arrival date. The varying launch times on any given day produce a negligible effect on the characteristics of the heliocentric trajectory.

The trajectory characteristics for the near planet period are determined by the launch and arrival dates and by the selected inclination, periapsis and apoapsis altitudes, and the amount of apsidal rotation of the elliptical orbit.

3. Trajectory Requirements

a. General

Two spacecraft are to be launched on close flyby trajectories to Mars in 1973 such that the spacecraft can retro into near synchronous elliptical orbits around the planet. The launches will be conducted from a single launch pad utilizing the Titan/Centaur launch vehicle.

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b. Trajectory Type

Only Type I trajectories (those which have heliocentric transfer angles less than 180°) shall be used. These trajectories require lower flight times, approach speeds, and communication distance at arrival than the Type II trajectories (which have transfer angles greater than 180°).

c. Ascent Mode

The ascent mode from launch to injection will utilize the parking orbit. A tentative parking orbit of 90 n.m. is being assumed, but must be examined in detail for spacecraft heating considerations.

The minimum parking orbit coast times will be between about 20 to 60 minutes for all trajectories under consideration. Final injection location will take place roughly over the Indian Ocean for southeast launches and in the Pacific Ocean for northeast launches (see Figure 5B-1).

The direct ascent mode cannot be used in 1973 because of the significant payload losses which result.

d. Launch Azimuth

The total launch azimuth corridor to be used in 1973 will be determined by the exact limits allowed by range safety. In addition, the azimuth corridor will be constrained by the geometrical properties of the trajectories as well as tracking coverage considerations.

Figure 5B-2 shows the geometrical restriction of launch azimuth from Cape Kennedy as a function of declination of the departure asymptote. The values of the asymptote declination in 1973 depend on launch and arrival dates and vary between about +50° to +20°.

For planning purposes, the entire azimuth corridor of 45° to 115° is assumed available. Only a small sector of the corridor will be used on any given launch date. The exact corridor will be established after the range safety limits have been established and additional trade-offs performed. The extreme northeast launch azimuths are desirable to increase the total length of the launch period (see section V.B.3.f). However, their

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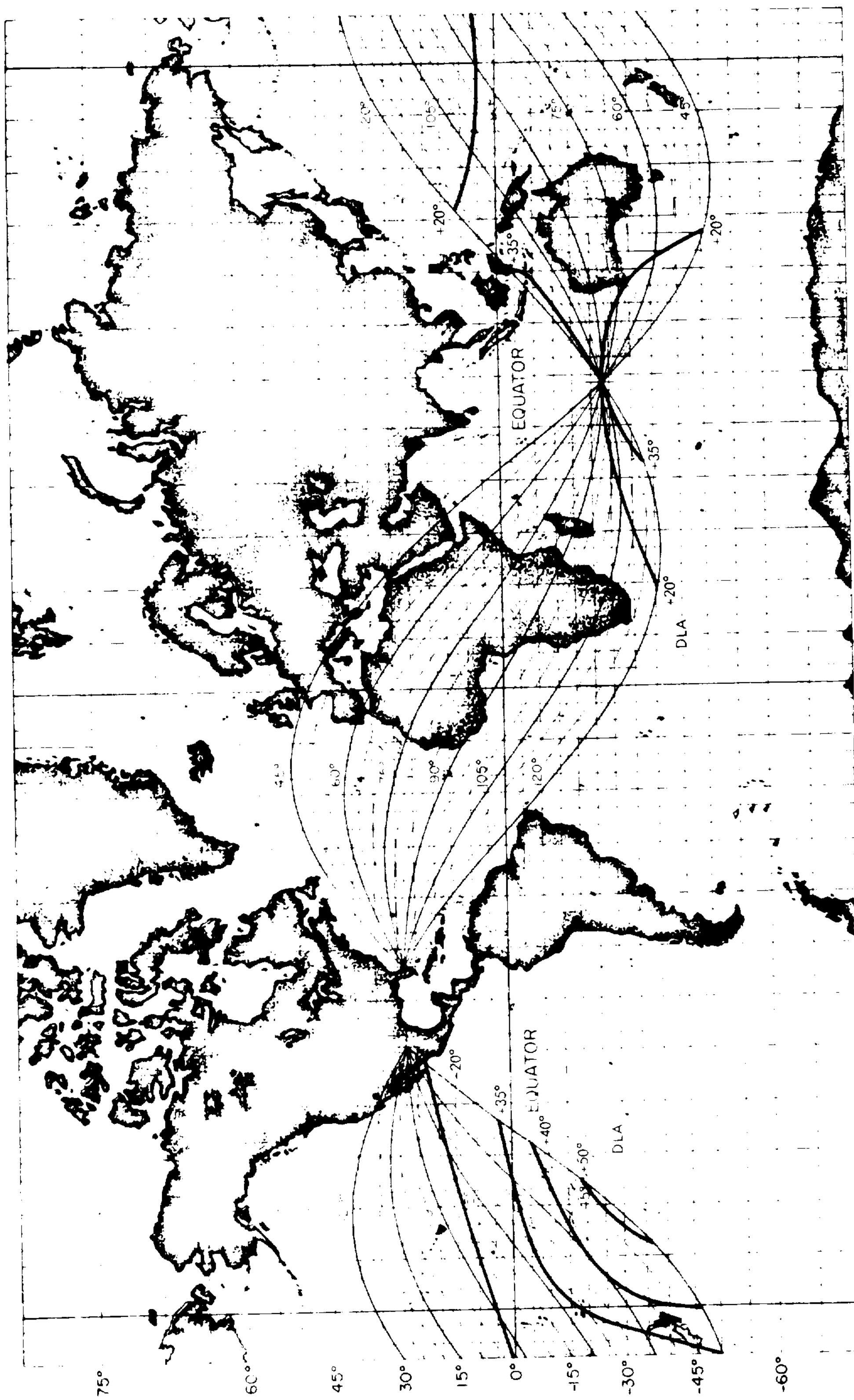

Figure 5B-1. Final Injection Locations for Various Departure Asymptote Declinations

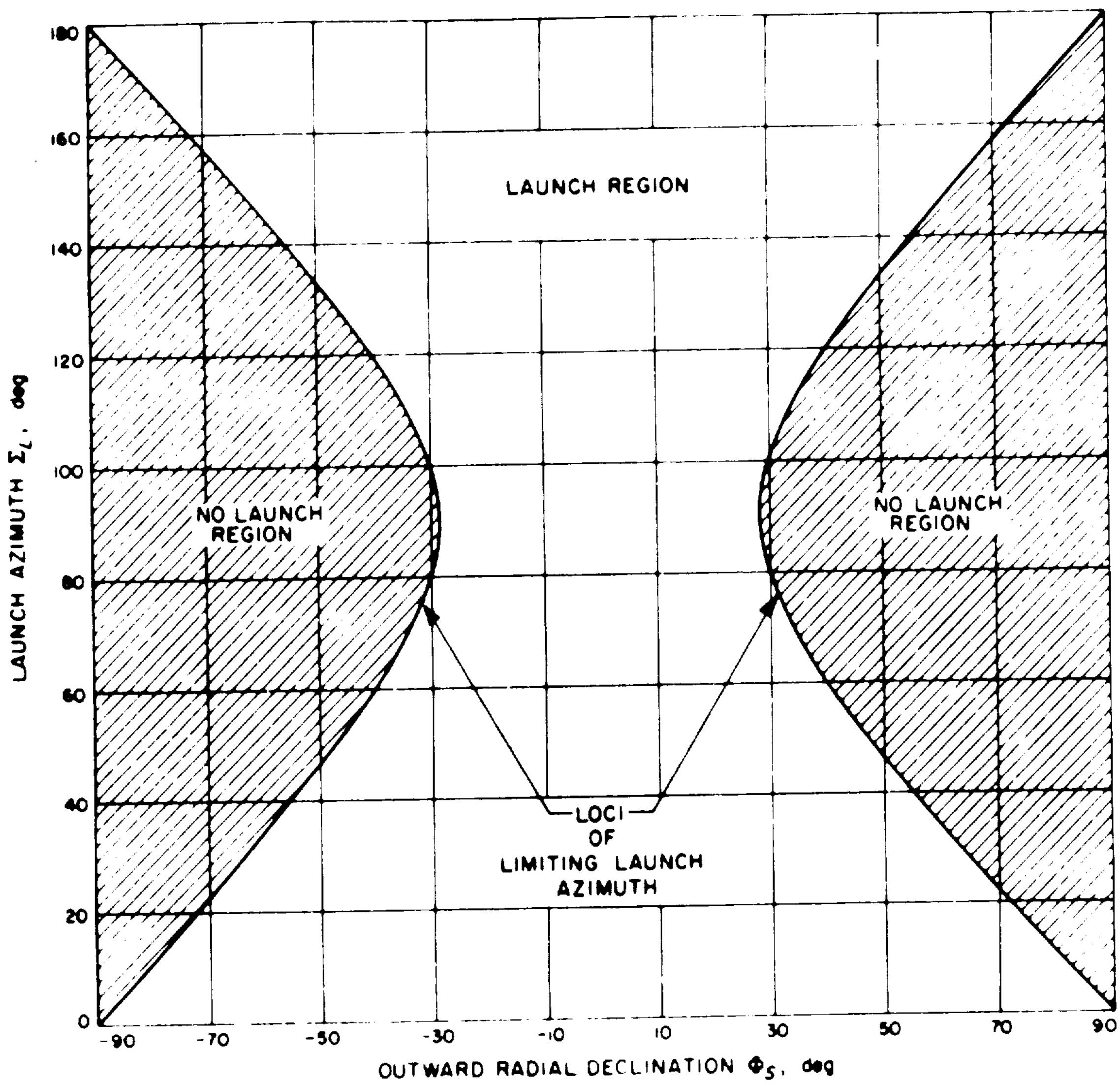

Figure 5B-2. Limiting Launch Azimuth vs. Departure Asymptote Declinations for Cape Kennedy acceptability by range safety is in doubt since the launch vehicle will fly over populated land masses in Europe (see Figure 5B-1).

e. Launch Windows

The daily launch window will depend on the exact launch azimuth and corridor. A minimum daily window of 30 minutes is probably acceptable.

f. Launch Period

The beginning of the launch period will depend on the exact arrival date and will be restricted by the maximum allowable declination of the departure asymptote. This maximum declination will be dictated by the minimum launch azimuth allowed by range safety (for northeast launches) or by the maximum allowed launch azimuth (if southeast launches are used). For an azimuth limit of 45°, the asymptote declination must be less than about 50°; for an azimuth limit of 65° or 115°, the declination must be less than about 36° (see Figure 5B-2).

The end of the launch period will also depend on the exact arrival date and be restricted by the payload capability, or thus the geocentric energy C_3. Estimates show that the maximum allowable C_3 for a 7335-lb spacecraft is about 20.9 km^2/sec^2, using a 45° launch azimuth. This is shown in section V.F. An additional small increase in C_3 can be accomplished by increasing the launch azimuth (C_3 = 22.7 for azimuth of 65° or 115°).

The length of the total launch period for the current mission design is around 40-45 days using the extreme launch azimuth of 45°. This launch period may be cut by as much as 20 to 25 days if the launch azimuth limit is 65° or 115°.

g. Arrival Dates

The arrival date will be restricted by the hyperbolic approach speed, the desired landing site latitudes of the capsule, the desired inclination of the orbiter, and by any communication distance limitations imposed by the lander or orbiter.

Currently the hyperbolic approach speed must be kept less than about 3.3 km/sec for the conceptual spacecraft design and orbit design of section V.C.

h. Launch and Arrival Date Corridor

Figure 5B-3 shows the permissible launch and arrival date corridor that appears consistent with the previously mentioned guidelines.

The two traces for spacecraft 1 and 2 indicate the launch and arrival dates of the two spacecraft to obtain an orbital inclination of 40° and to land the capsules at Martian latitudes of 10° and 20° north, respectively, after 10 days in orbit. The requirements which can be satisfied are: the capsule can land at an angle of 30° from the evening terminator; the true anomaly of the impact point can be -15° (PER); apsidal rotations are allowed within the orbit insertion impulsive ΔV allocation of 1350 m/sec. The two traces will vary as the above parameters (e.g., orbital inclination, landing site latitude, time in orbit before capsule separation, etc.) are varied.

i. Arrival Separation

Because of mission operation considerations, the arrivals of the two spacecraft should be separated by an adequate margin. Preferably, the final lander evaluation should be obtained before the final de-boost into initial orbit for the second spacecraft.

For planning purposes the arrival separation will be assumed greater than the time that the first lander is kept in orbit (a maximum of 30 days) plus 5 days of lander evaluation.

j. Arrival Times

Arrival at Mars (orbit insertion) will occur over a 210-ft DSN tracking station. The current guideline is that three DSN 210-ft antennas will be available to support the Viking mission. Thus, the spacecraft will be in view of at least one station at any time of the day. Therefore, the exact arrival time will be selected on the basis of landing site surveillance and longitudinal accessibility.

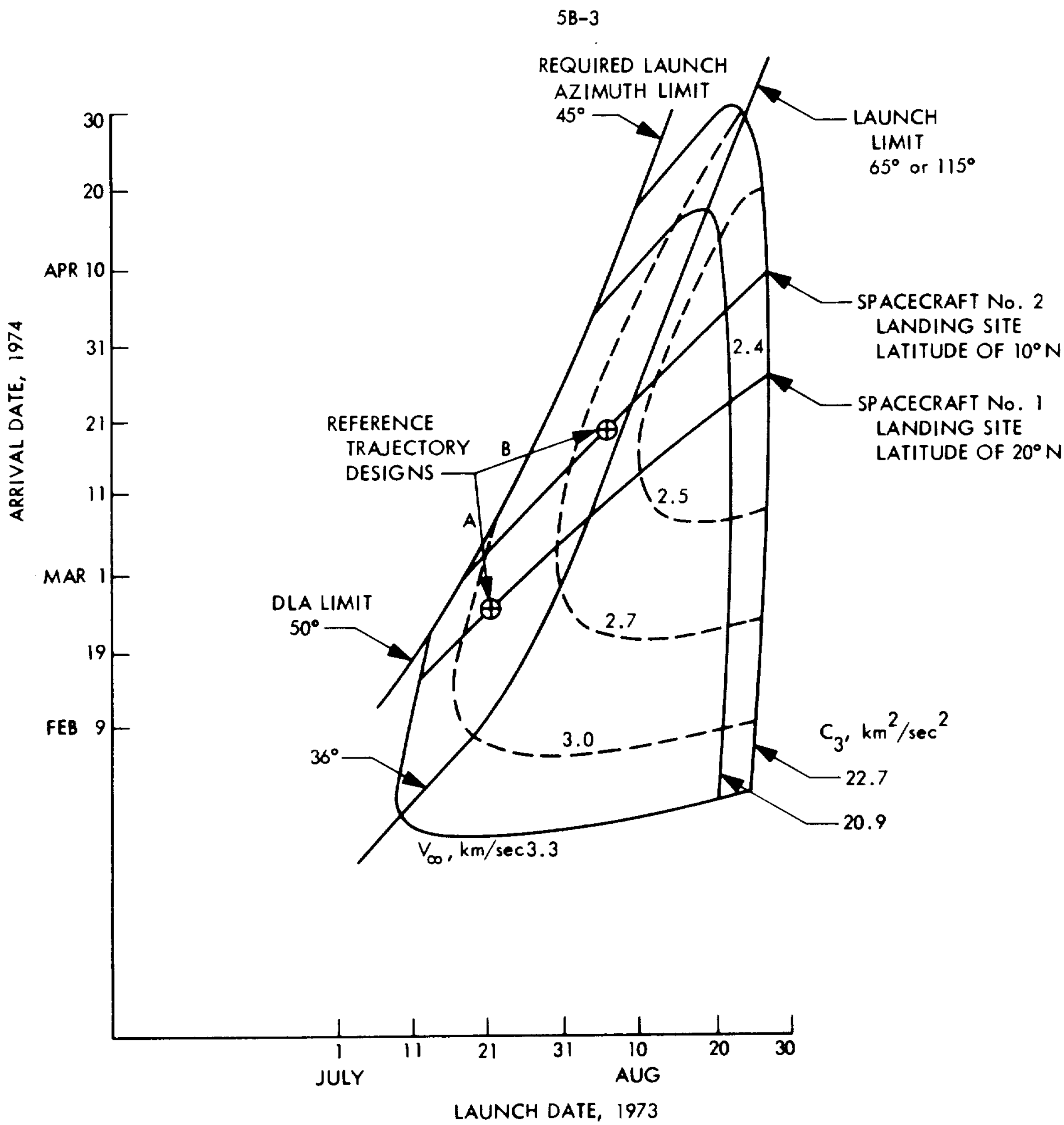

Figure 5B-3. Launch and Arrival Date Corridor

4. Baseline Trajectories

Two baseline trajectories, called A and B in Figure 5B-3, were given to the orbiter design team for detailed studies. Some of the characteristics of these two trajectories are given in Table 5B-1.

a. Mars Approach

Figures 5B-4 and 5B-5 show the view of Mars for the two trajectories as seen from the approach asymptote of the hyperbolic orbit. The plane of the paper is defined to be the aiming plane for the hyperbolic trajectory. The edge view of the desired elliptical orbits are also shown (for orbital inclination of 40°).

b. Orbit Insertion

Table 5B-2 gives the characteristics of the minimum impulsive transfers into the desired elliptical orbits. The actual burn will take up to 45 minutes and will be approximately symmetrical about the impulsive transfer point.

c. Earth Related Information

Figures 5B-6 and 5B-7 show the station views of the spacecraft on the arrival dates of the two baseline trajectories. Figure 5B-8 shows the Earth-to-Mars communication distance as a function of time.

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Table 5B-1. General Baseline Trajectory Characteristics

TABLE 5B-1. GENERAL BASELINE TRAJECTORY CHARACTERISTICS

	C_3 $\frac{Km^2}{sec^2}$	LAUNCH DATE 1973	ARRIVAL DATE 1974	COMMUNICATION DIST AT ARRIVAL, millions of Km	DEPARTURE ASYMPTOTE DECLINATION, deg	APPROACH SPEED, Km/sec	LAT OF VERTICAL IMPACT POINT, deg	LAT OF SUB-EARTH POINT ON MARS, deg	LAT OF SUB-SOLAR POINT ON MARS, deg
TRAJECTORY A	17.1	7/21	2/25	206.5	43.7	2.937	+15.1	-13.0	+3.5
TRAJECTORY B	17.1	8/5	3/20	239.2	40.3	2.658	+15.5	-7.5	+7.5

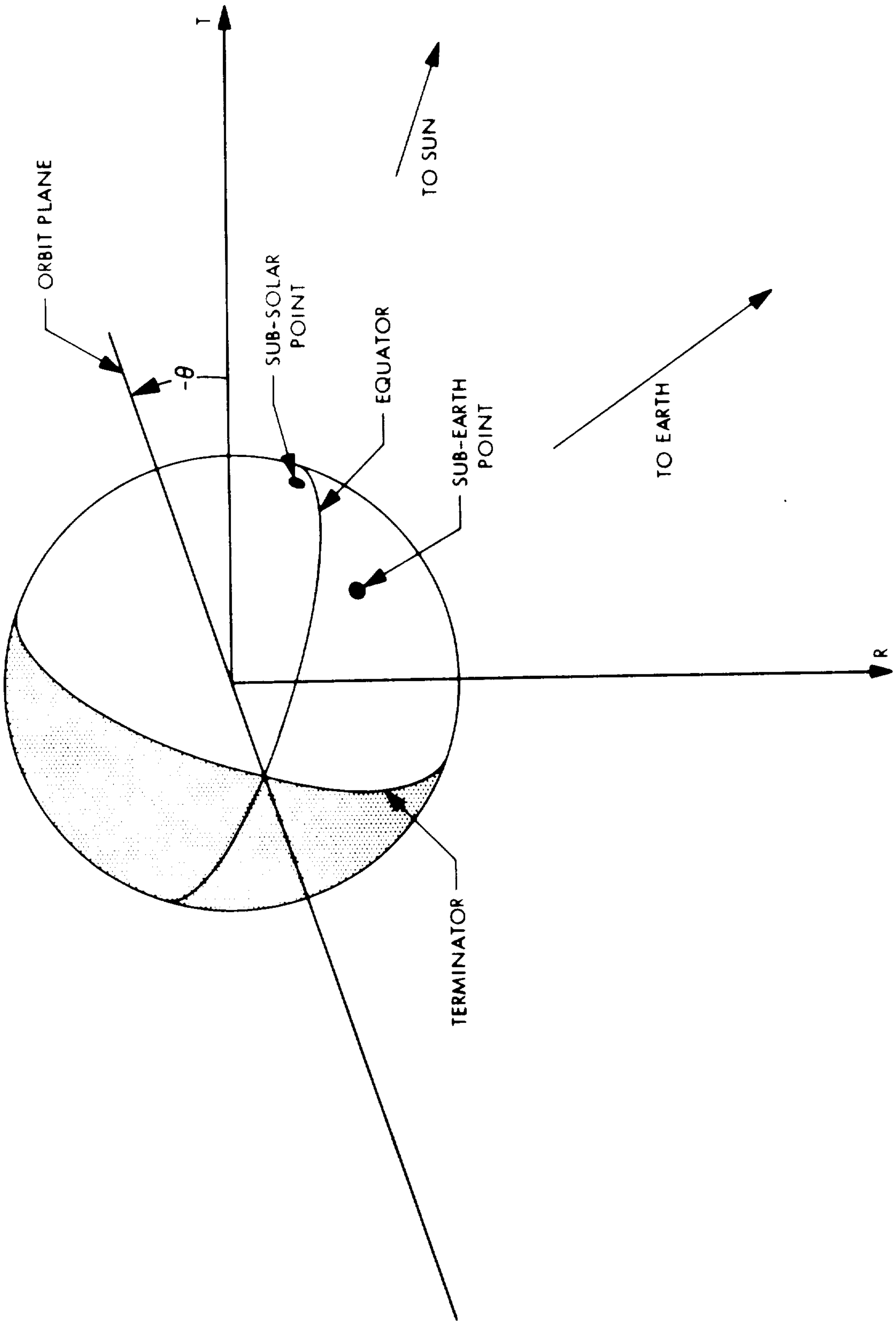

Figure 5B-4. View of Mars from Approach Asymptote, Trajectory A

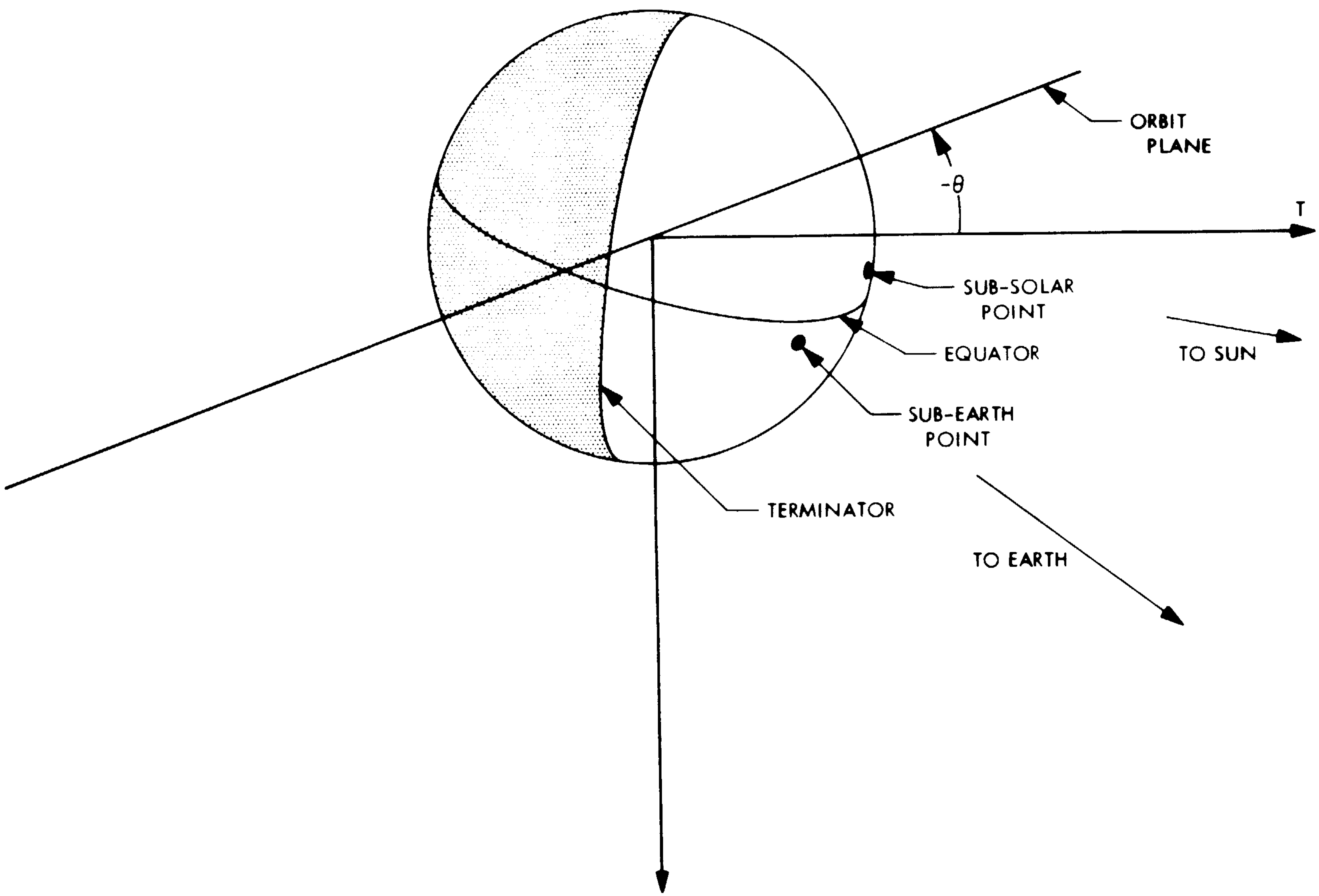

Figure 5B-5. View of Mars from Approach Asymptote. Trajectory B

Table 5B-2. Arrival Characteristics of Baseline Trajectories

	INC TO EQUATOR, deg	HYPERBOLIC PERIAPSIS ALT, Km	HYPERBOLIC TRUE ANOMALY OF INSERTION, deg	APSIDAL ROTATION, OF ELLIPSE, deg	ELLIPTICAL PERIAPSIS ALTITUDE, Km	ELLIPTICAL TRUE ANOMALY OF INSERTION, deg	ORBIT INSERTION ΔV, Km/sec	CONE ANGLE OF INSERTION ΔV, deg	CLOCK ANGLE OF INSERTION ΔV, deg
TRAJECTORY A	40	≈1150	≈-15	+11.2	1000	≈-25	1.15	≈ 52	≈ 213
TRAJECTORY B	40	≈1700	≈-38	+25.0	1000	≈-65	1.14	≈ 72	≈ 225

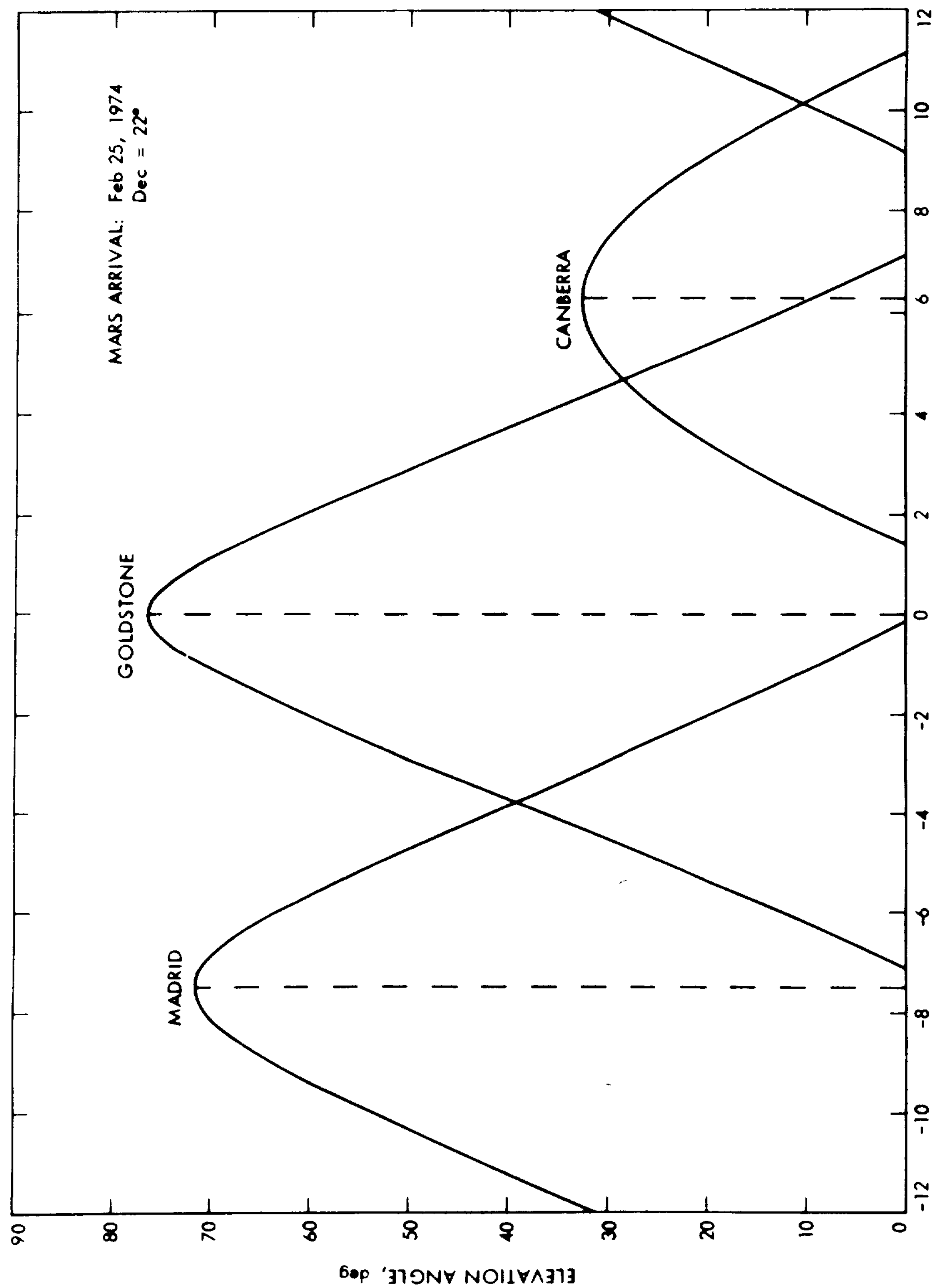

Figure 5B-6. Station View of Trajectory A at Arrival

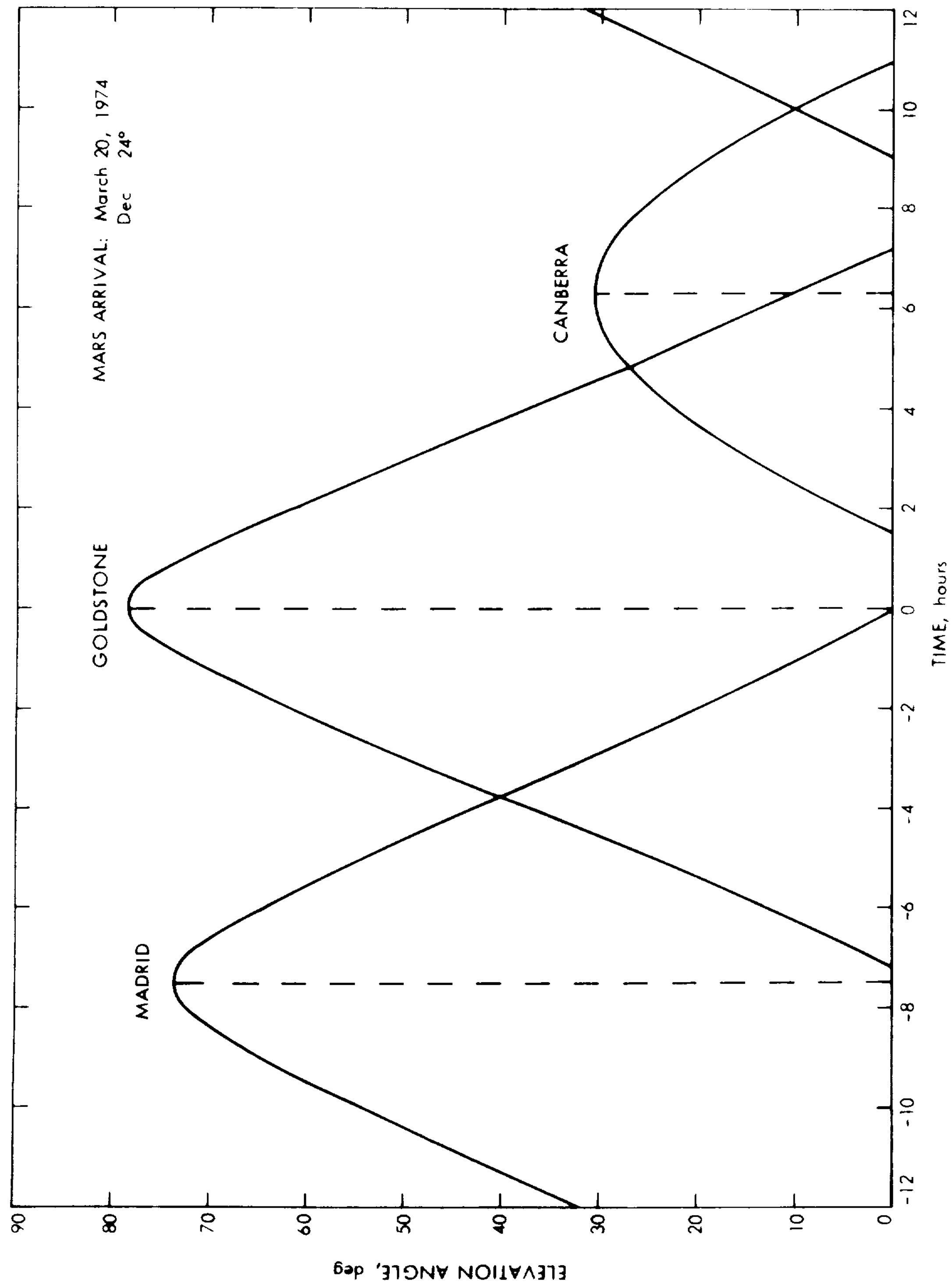

Figure 5B-7. Station View of Trajectory B at Arrival

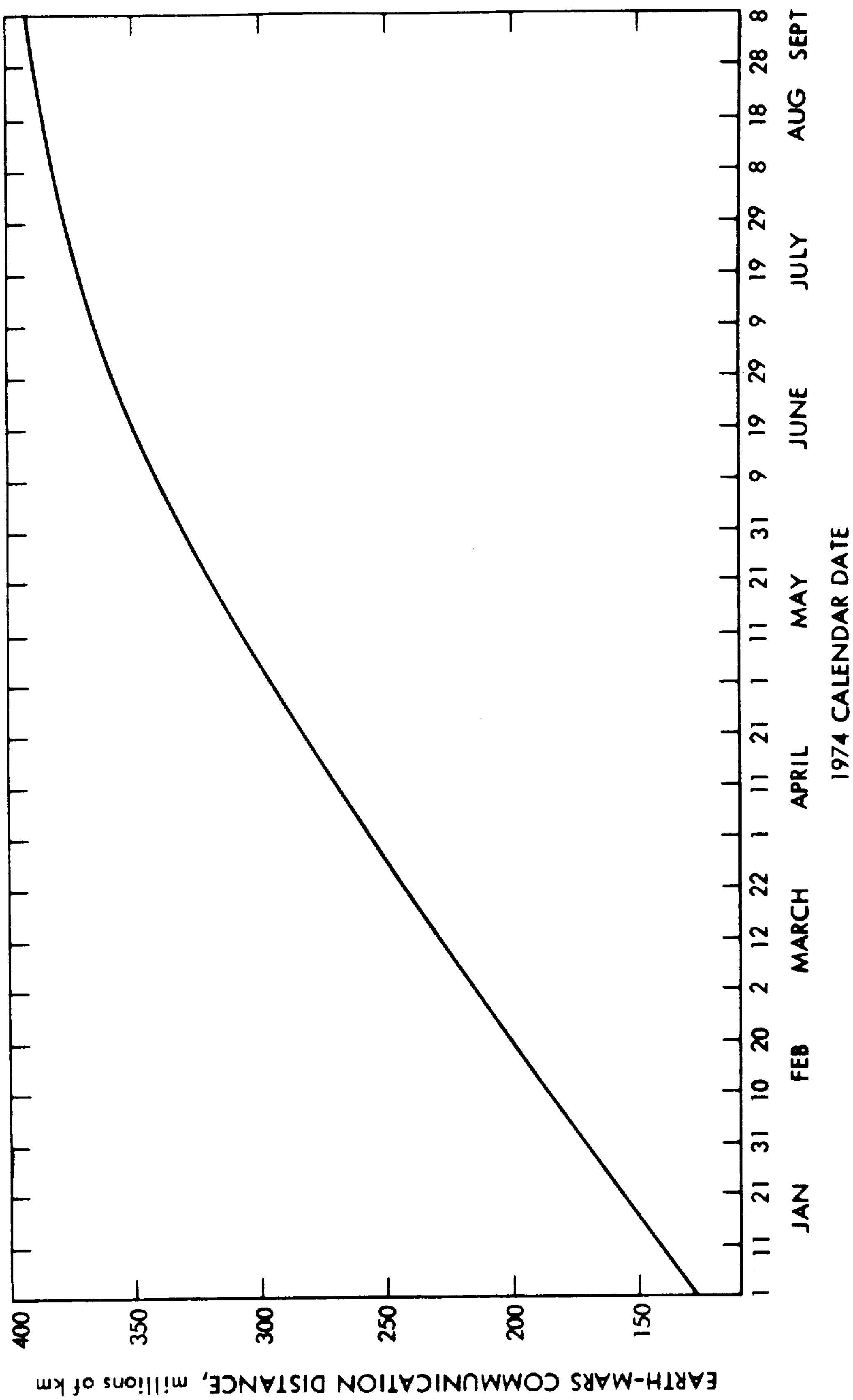

Figure 5B-8. Earth-Mars Communication Distance

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C. ORBIT SELECTION AND CHARACTERISTICS

1. Requirements on the Orbiter

The orbiter must satisfy two primary requirements which infleunce the orbit selection. These are to provide support for the lander and perform scientific reconnaissance of the Martian surface. In the ensuing discussion, the orbit is divided into phases, the purpose of each phase being to meet the requirements placed on the orbiter which are peculiar to that phase. Each of the orbiter requirements can be sub-divided as follows:

a. Lander Support

(1) Landing site surveillance -- The orbiter is to provide TV and IR surveillance of all potential landing sites over all longitudes.

(2) Capsule delivery -- The capsule will be deflected out of a synchronous orbit. The orbit must be trimmed to Mars synchronous, and the capsule must be released at the proper time and in the proper direction to ensure favorable entry geometry (i. e., controlled flight path angle at atmospheric entry, acceptable post-landed view from orbiter on landing pass, etc.).

(3) Lander relay -- The orbiter is to provide a real-time data relay link between the lander and Earth during capsule approach and atmospheric entry and for as long after touchdown as is possible on the first pass. The relay link must be re-established on the next three passes over the lander. Subsequent to this initial relay period the orbiter will perform a trim maneuver to de-synchronize its orbit, but not before the direct telecommunication link has been established between the lander and Earth. After the orbit is de-synchronized, the orbiter is required to pass over the lander and re-establish the relay link once a month for the next 90 days.

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b. Scientific Reconnaissance of Planet Surface

After the inital lander-orbiter-Earth relay period, the orbiter will change its orbital period so as to survey the planet over all longitudes. Its sole lander support function will be to provide a real-time data link at monthly intervals. However, should the lander direct link fail, the orbiter must then position itself in synchronous orbit which passes over the lander and provide a data relay link for the lander.

2. Orbit Constraints

The orbit is constrained in two ways. They are: spacecraft energy limitations and spacecraft engineering requirements.

a. Energy Constraints

Energy constraints are those which bound the orbit selection process by defining the maximum variation in orbital elements which is within the orbiter's energy capability. The orbiter will have velocity capability sufficient to transfer from a hyperbolic approach trajectory into a near-synchronous orbit, plus the capability to perform several trim maneuvers.

(1) Inclination angle -- Since the energy required to change the orbital plane is high, this type of maneuver is ruled out. Thus, orbit inclination angle is constrained to be no smaller than the angle between the incoming asymptote and Mars' equator. So, for a spacecraft arriving February 25, 1974, $i \geq 16.1$ deg, and for a spacecraft arriving March 30, 1974, $i \geq 15.9$ deg.

(2) Apsidal rotation -- The transfer from hyperbolic to elliptic orbit is accomplished with minimum energy if the transfer is performed at hyperbolic periapsis to elliptic periapsis (so-called periapsis-to-periapsis transfer). The line of apsides of the elliptic orbit

can be rotated at the time of orbit insertion by performing the transfer at some point other than hyperbolic periapsis; this is illustrated in Figure 5C-1, where the angle APS is the apsidal rotation induced at the time of orbit insertion. If an apsidal rotation is induced at orbit insertion, the orbiter must pay in reduced velocity capability. So, although it is not possible at this time to place a hard constraint on APS, it can be said that there is a trade-off between APS and orbiter velocity capability. The velocity required to achieve an apsidal rotation is illustrated in Figure 5E-1, which shows minimum orbit insertion velocity as a function of APS for three approach velocities, 2.7. 2.85, and 3.0 km/sec.

b. Engineering Constraints

Engineering constraints are those which bound the orbit selection process so as to ensure that various engineering requirements are met. The engineering constraints on the orbit apply to occultation of the Sun and Earth as seen by the orbiter.

(1) Sun occultation -- The orbit must be such that the Sun is not occulted until the capsule is separated from the orbiter. This is necessitated by the many activities (e.g., orbit insertion, orbit trims, landing site surveillance, capsule separation) which require battery power. After capsule separation, the battery constraint will relax and the orbiter can tolerate successive Sun occultations of approximately 3 hours duration. Figure 5C-2 shows the period of Sun occultation as a function of the number of days in orbit for the two reference trajectories. Both orbits are inclined 40 degrees to Mars' equator.

(2) Earth occultation -- The orbit must be such that the

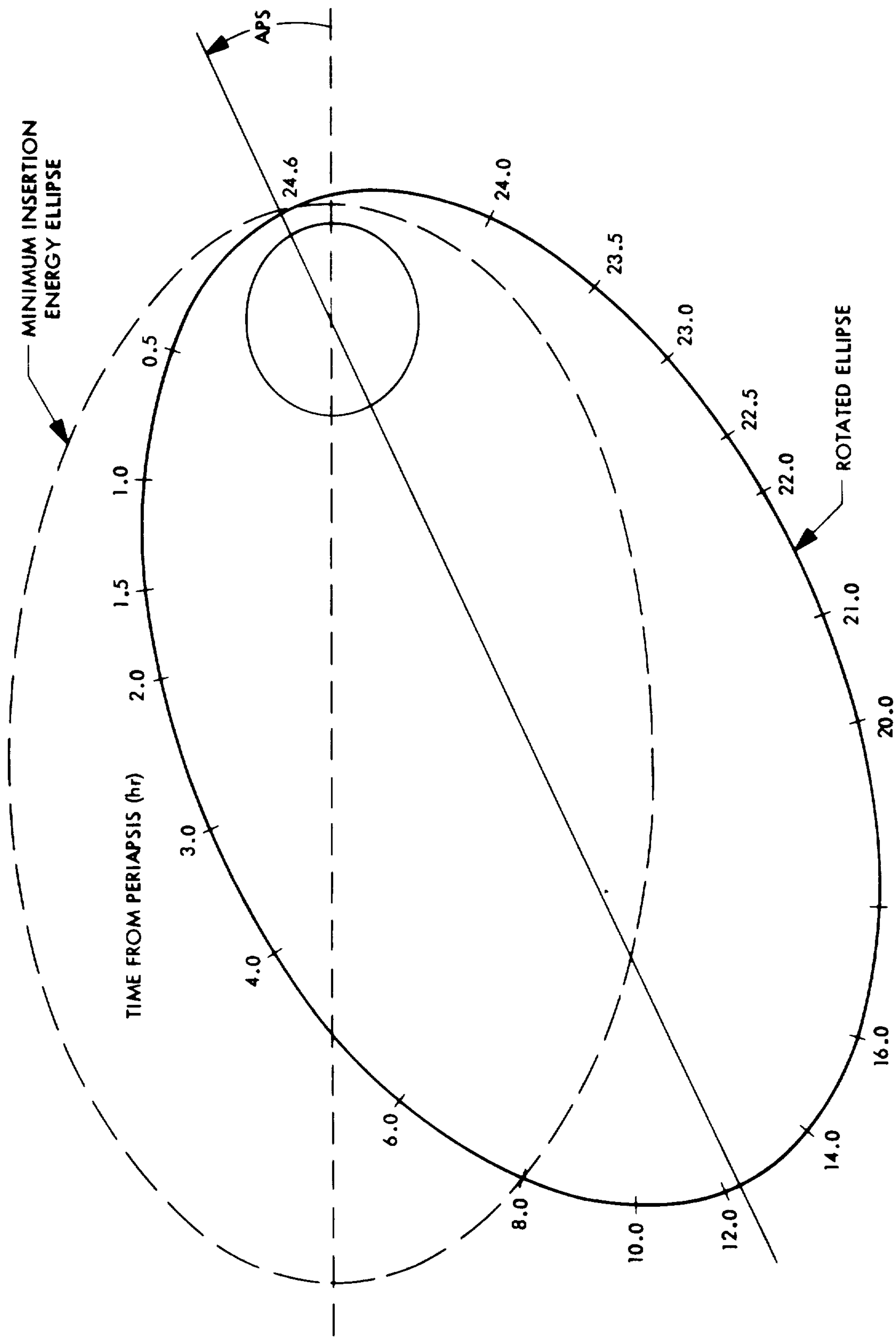

Figure 5C-1. Apsidal Rotation at Orbit Insertion

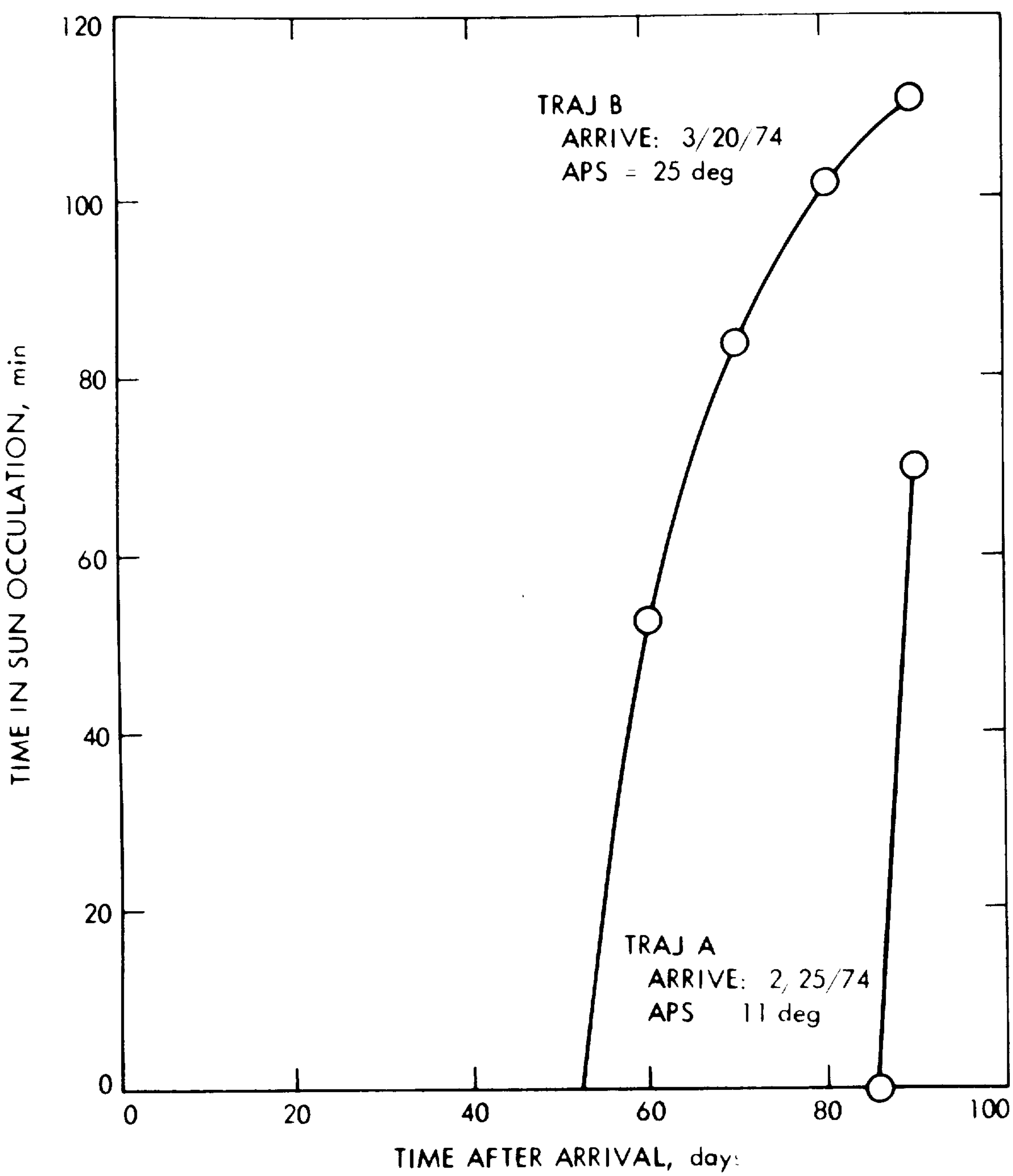

Figure 5C-2. Period of Sun Occultation as a Function of Number of Days in Orbit

Earth is not occulted during any maneuvers which require that the orbiter roll axis be off the Sun line.

3. Orbit Characteristics

The general satellite orbit parameters are shown on Figure 5C-3. Periapsis altitude, h_p, is a flexible parameter; however, the bulk of the analysis to date is based on a periapsis altitude of 1000 km. The requirements placed on the orbiter dictate the following orbit characteristics.

a. General

(1) Orbital period -- The spacecraft must be inserted into a slightly non-synchronous orbit to allow landing site surveillance over all longitudes. The actual period that is selected will depend on the time period which it is desired to devote to scanning 360° in longitude. Figure 5C-4 shows the required variation in period from synchronous as a function of time to cover the planet.

The period must be synchronized before the capsule is separated, and it must remain synchronous for at least three succeeding revolutions. Mars' period of rotation is 24.623 hours; if Mars were a perfect sphere, this would be the period to synchronize the orbiter with the surface. However, because Mars is an oblate spherioid, the longitude of the ascending node (Ω) regresses (i.e., it moves westward). This is illustrated in Figure 5C-5, which shows the nodal regression rate as a function of inclination angle. Hence, the period must be adjusted to account for nodal regression. Figure 5C-6 shows the period which synchronizes the orbit as a function of inclination angle. (Another effect of Mars' oblateness is that the argument of periapsis (ω) advances. Figure 5C-7 shows the apsidal precession rate as a

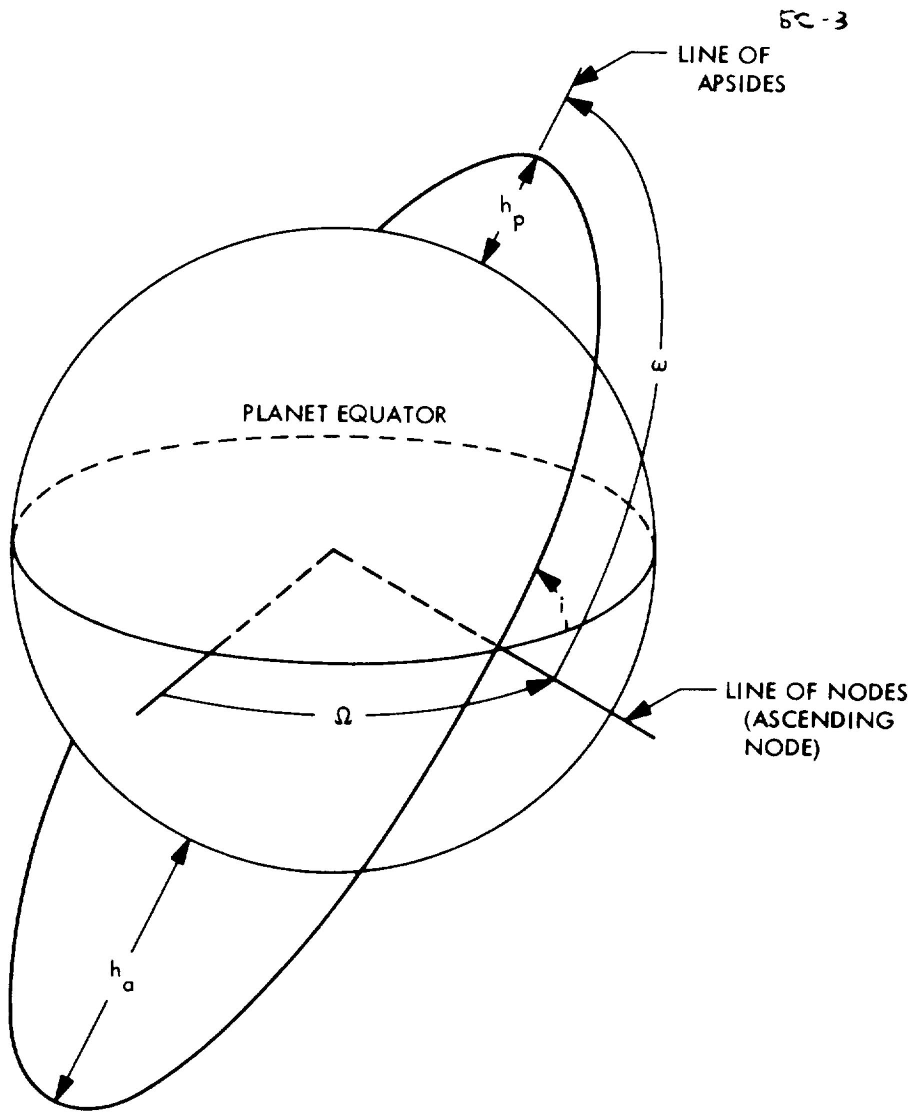

Figure 5C-3. General Satellite Orbit Parameters

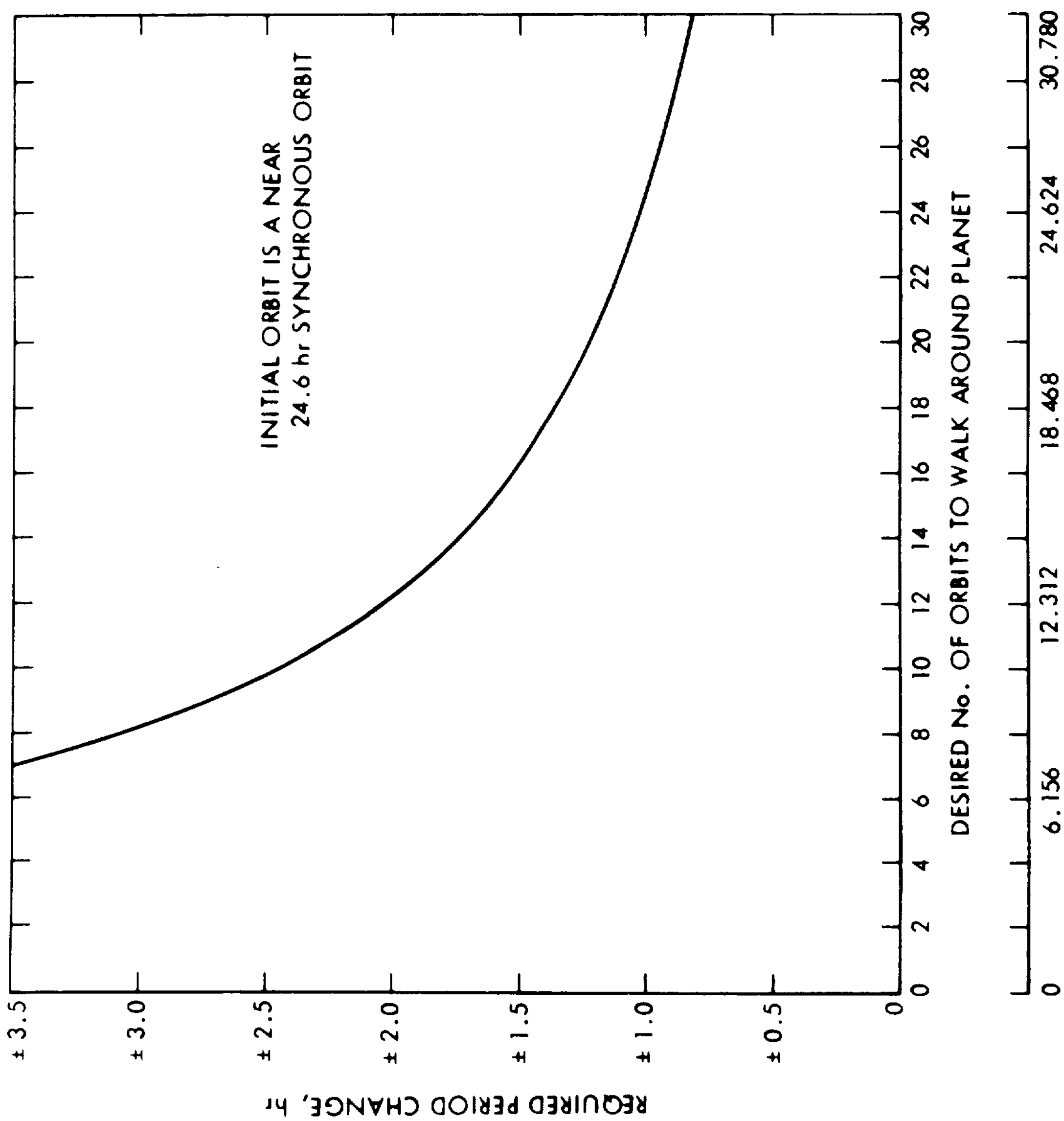

Figure 5C-4. Period Change Required to Traverse All Longitudes

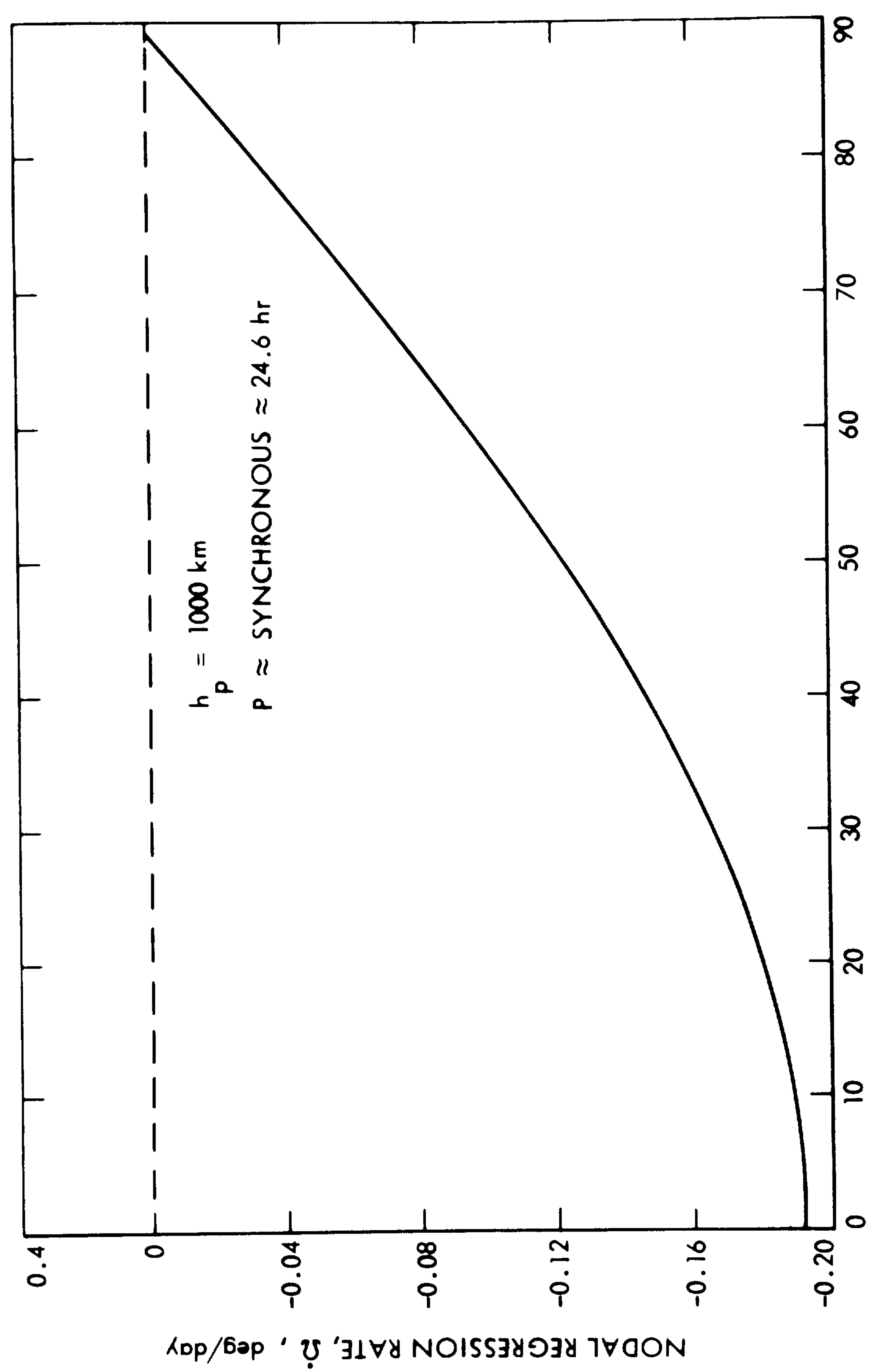

Figure 5C-5. Nodal Regression Rate as a Function of Inclination Angle

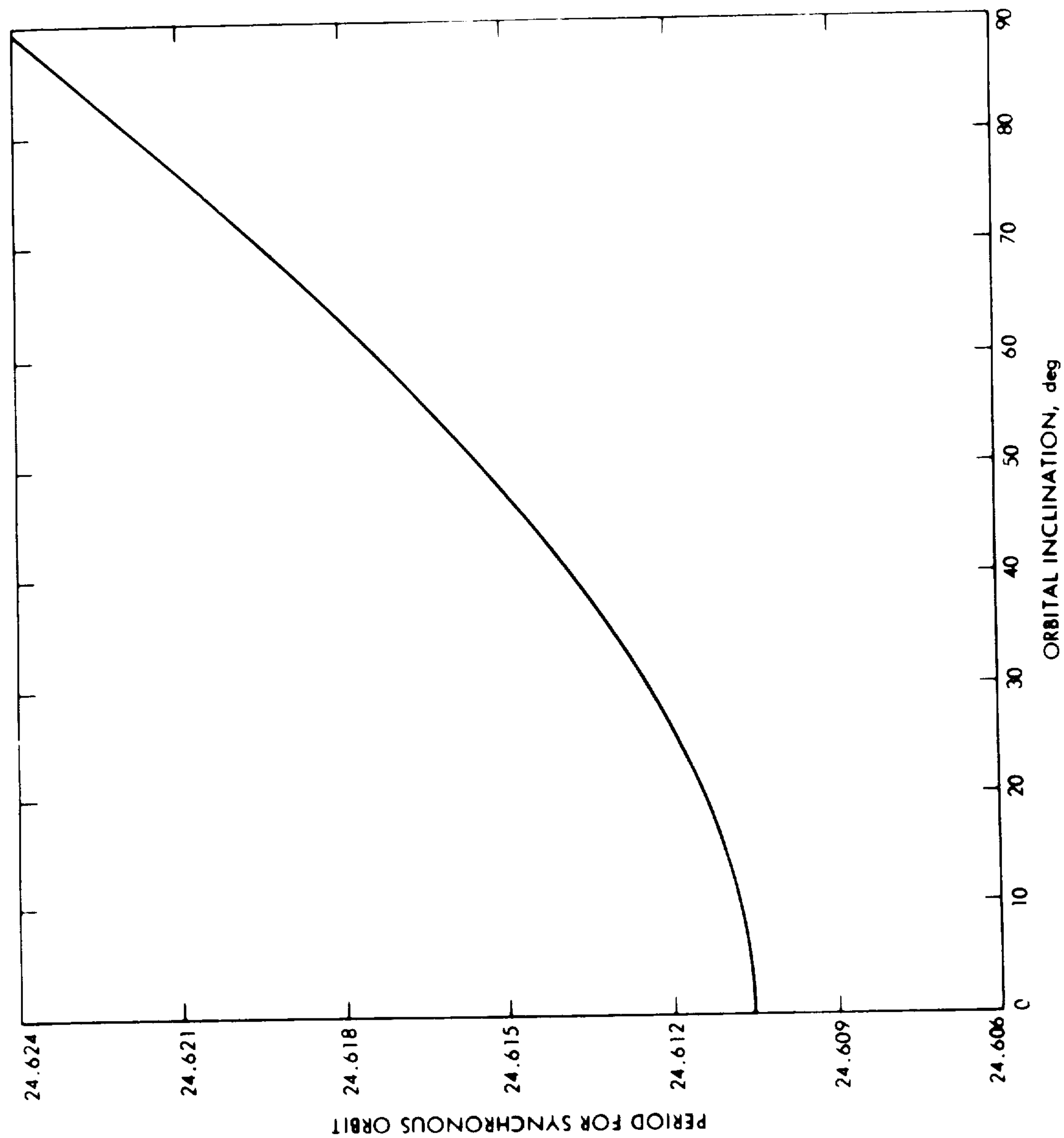

Figure 5C-6. Period for Synchronous Orbit over Landing Site

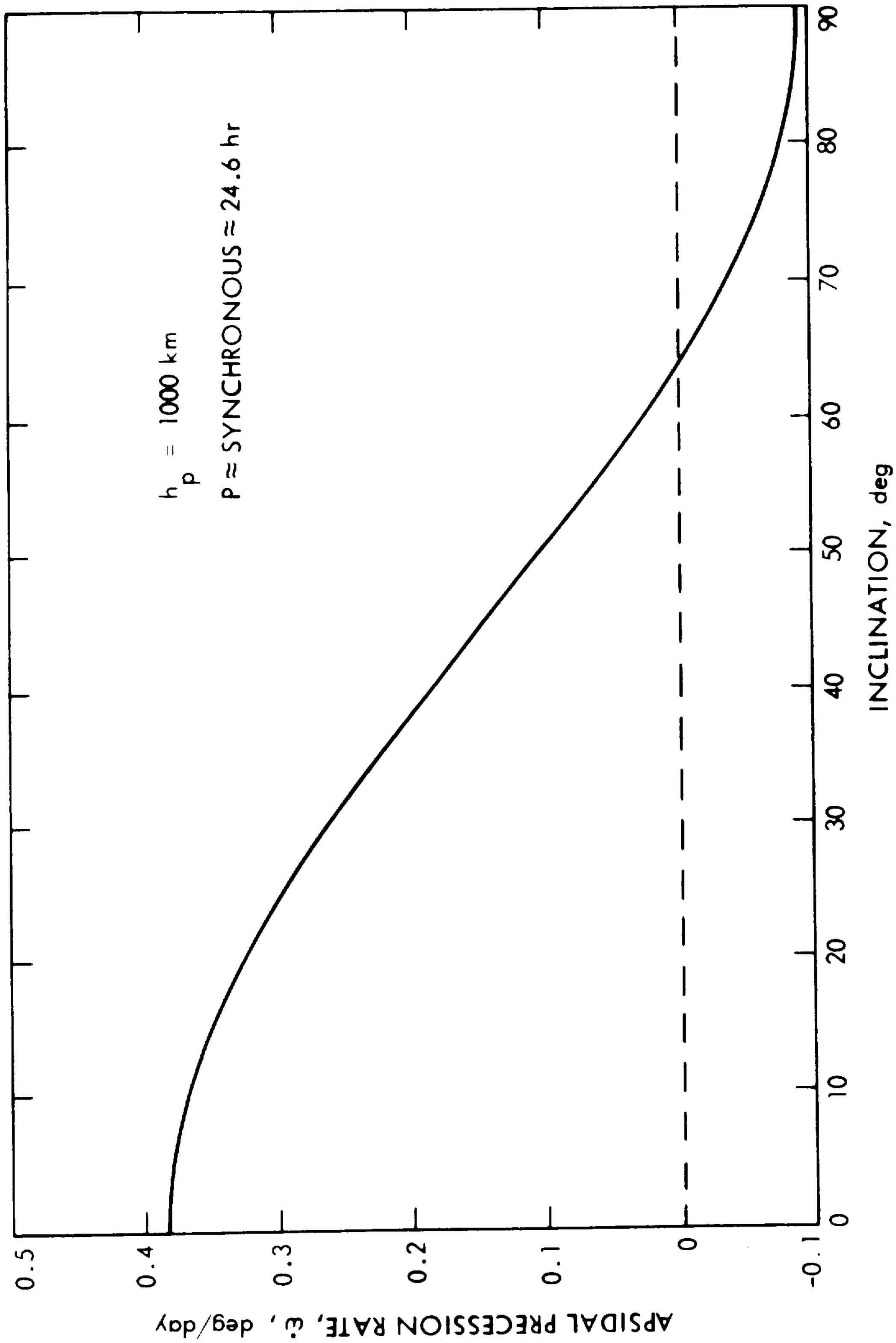

Figure 5C-7. Apsidal Precession Rate as a Function of Inclination Angle function of inclination angle for a synchronous orbit with periapsis altitude of 1000 km).

To go into the scientific reconnaissance phase, the period must be adjusted to allow full longtudinal coverage of the planet. The orbiter is required to pass over the lander once a month for the next 90 days. From Figure 5C-4, a period adjustment of approximately ±.75 hr from synchronous would cause the orbiter to return in about 30 days.

The initial period is defined by the energy removed from the spacecraft trajectory at the time of orbit insertion. This subject, plus a discussion of the velocity required to change the period, are presented in Section 5E.

(2) Inclination angle -- The choice of orbital inclination is governed by the somewhat competing desires to perform scientific reconnaissance over much of the planet surface and to land and support the capsule in the latitude band between 0 and 30 degrees North. The former desire suggests high inclinations to cover a wide range of latitudes, while the latter desire suggests lower inclinations to increase dwell time over the lower latitudes and thus improve chances of placing the lander in a desired region under favorable conditions (lighting, etc.).

The orbital inclination is defined at the time of orbit insertion by selection of θ, the aim point polar angle in the R-T plane.* Table 5C-1 shows the relationship

* The R-T plane and the aim point in the plane are illustrated in Figures 5B-4 and 5B-5.

between θ, the aim point polar angle (measured clockwise from the T-axis), and orbital inclination for the arrival dates corresponding to the two reference trajectories.

b. Implications of Capsule De-Orbit, Approach, and Entry

Figure 5C-8 illustrates the orbit and de-orbit geometry. The angle PER is the central angle from spacecraft periapsis to the lander touchdown point at the time of touchdown, measured in the direction of spacecraft motion. B_{LOOK} is the angle from the local vertical at the lander to the orbiter. When the orbiter is approaching, B_{LOOK} is negative, and, when it is departing, B_{LOOK} is positive.

With these definitions in mind, attention is directed to Figures 5C-9 to 5C-13, which are plots of capsule flight path angle at atmospheric entry (altitude = 243.8 km) vs bus time, where bus time is the time from touchdown until the spacecraft passes directly over the lander, h_p is the periapsis altitude in km, and DELV is the capsule deflection velocity in meters/sec. Figures 5C-9, -10, and -11 are for h_p = 1000 km, and DELV = 100, 150, and 200 m/sec, respectively. Figures 5C-12 and -13 are for h_p = 2000 km, and DELV = 150 and 200 m/sec, respectively. Superimposed on Figures 5C-9 to -13 are contours of constant PER and constant ET (entry time), the time from deflection to atmospheric entry. The line labelled "B_{LOOK} = -75°" represents the constraint imposed to meet the requirement that $B_{LOOK} \geq -75$ deg at touchdown; any point to the right of this line violates this constraint. The line labelled "Minimum Post-Landed View" represents the constraint imposed to meet the requirement that the post-landed relay communication period be at least 12 minutes before the spacecraft passes over the lander's communication horizon; any point to the left of this line violates this constraint.* The plots for a periapsis altitude of 2000 km (Figure 5C-12 and -13) show the constraints for $B_{LOOK} \geq 60$ deg also. The "minimum post-landed view" constraint is not

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*There is nothing special about these constraints. They are flexible.

Table 5C-1. Relationship of Orbit Inclination to Aim Point Polar Angle θ

Inclination (degrees)	Arrival Date	θ (degrees)	First Node Crossed
60	2/25/74	-40.4	Descending
	3/20/74	-42.0	
50	2/25	-29.8	Descending
	3/20	-31.4	
40	2/25	-18.9	Descending
	3/20	-20.6	
30	2/25	-7.4	Descending
	3/20	-9.1	
20	2/25	6.2	Descending
	3/20	4.3	
16.1	2/25	18.2	Descending
15.9	3/20	16.7	
20	2/25	30.1	Ascending
	3/20	28.9	
30	2/25	43.8	Ascending
	3/20	42.4	

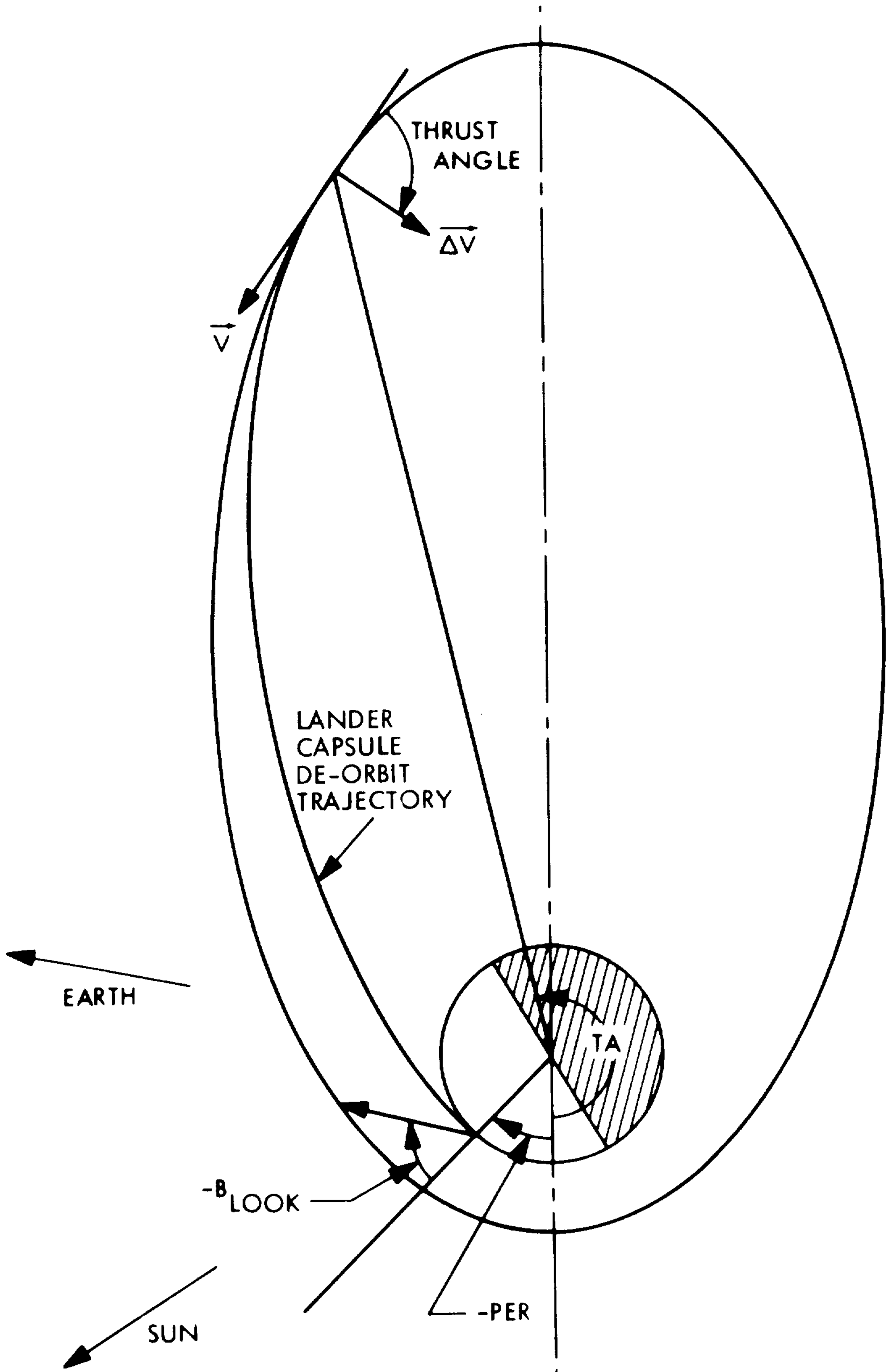

Figure 5C-8. Geometry of Orbit and De-Orbit Trajectory

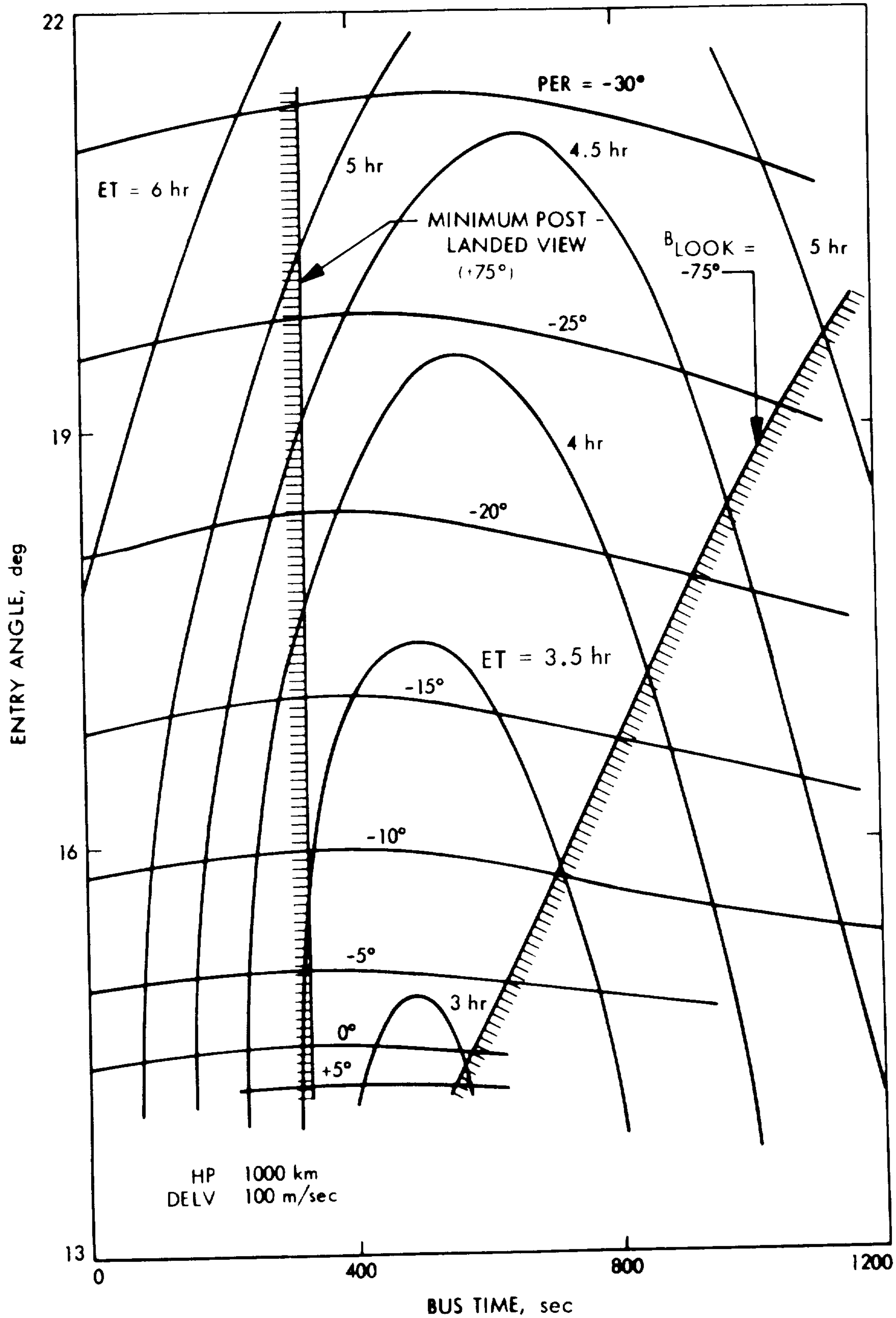

Figure 5C-9. Capsule Entry Design Chart, h_p = 1000 km, ΔV = 100 m/sec

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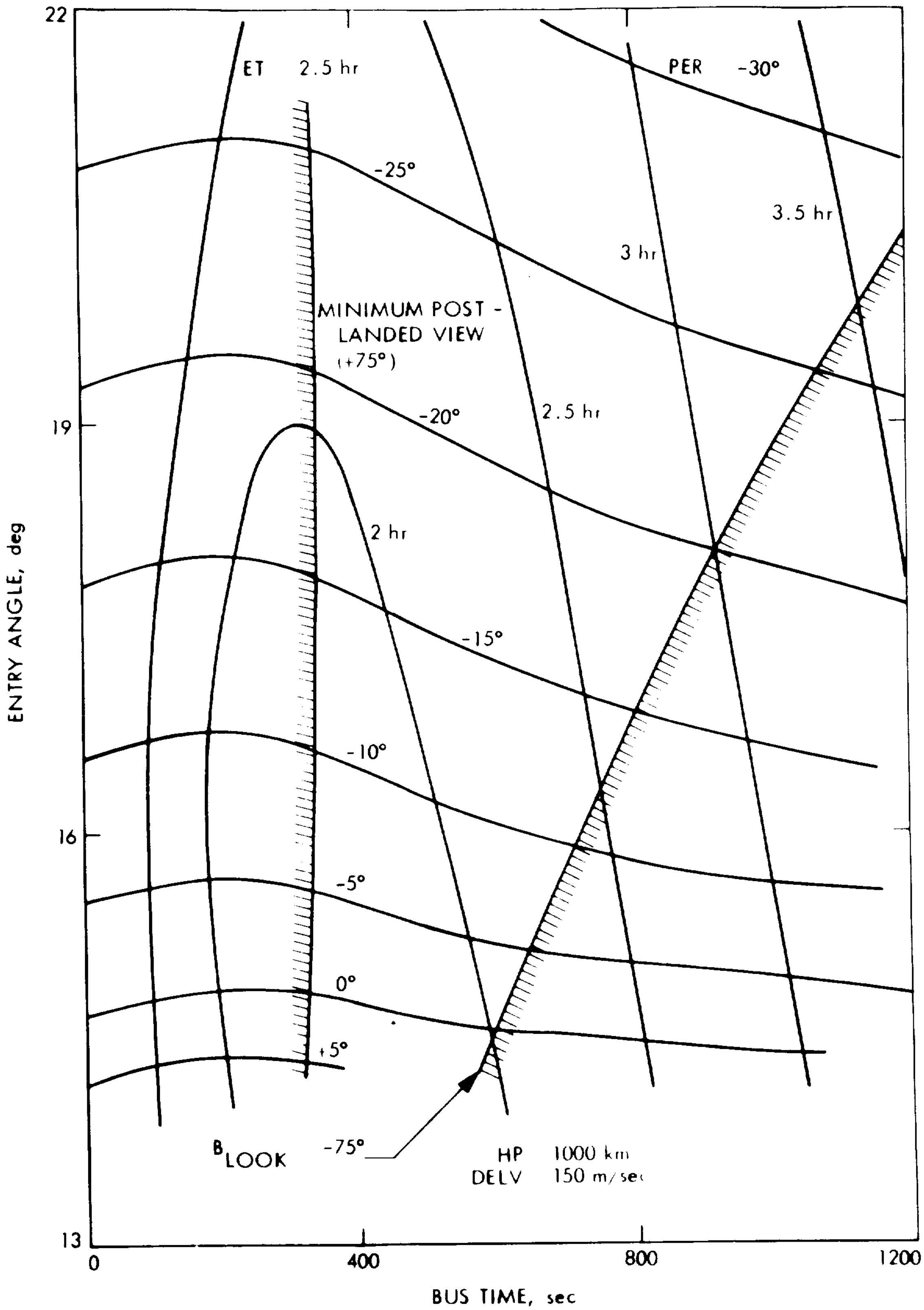

Figure 5C-10. Capsule Entry Design Chart, $h_p = 1000$ km, $\Delta V = 150$ m/sec

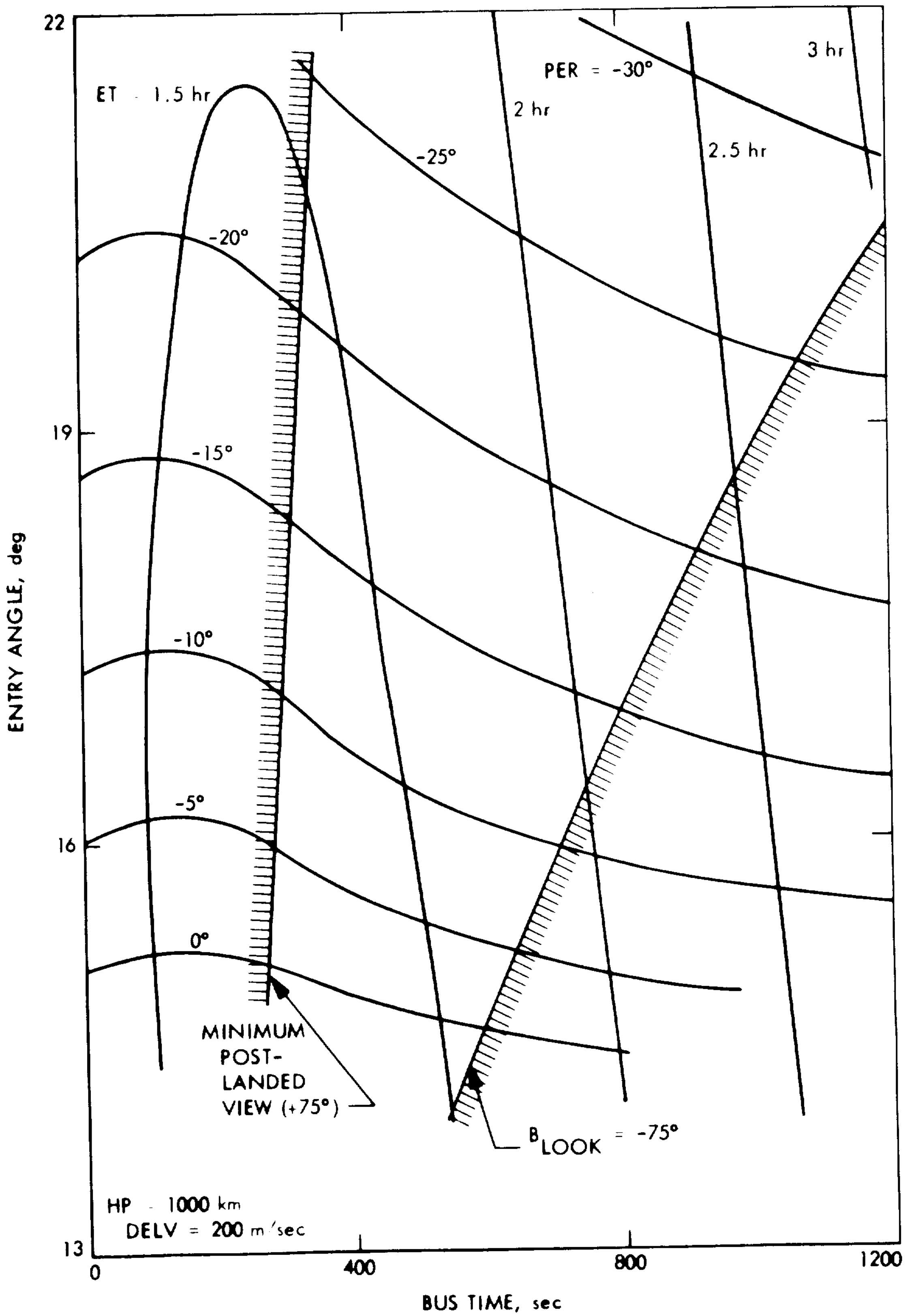

Figure 5C-11. Capsule Entry Design Chart, h_p = 1000 km, ΔV = 200 m/sec

C-2

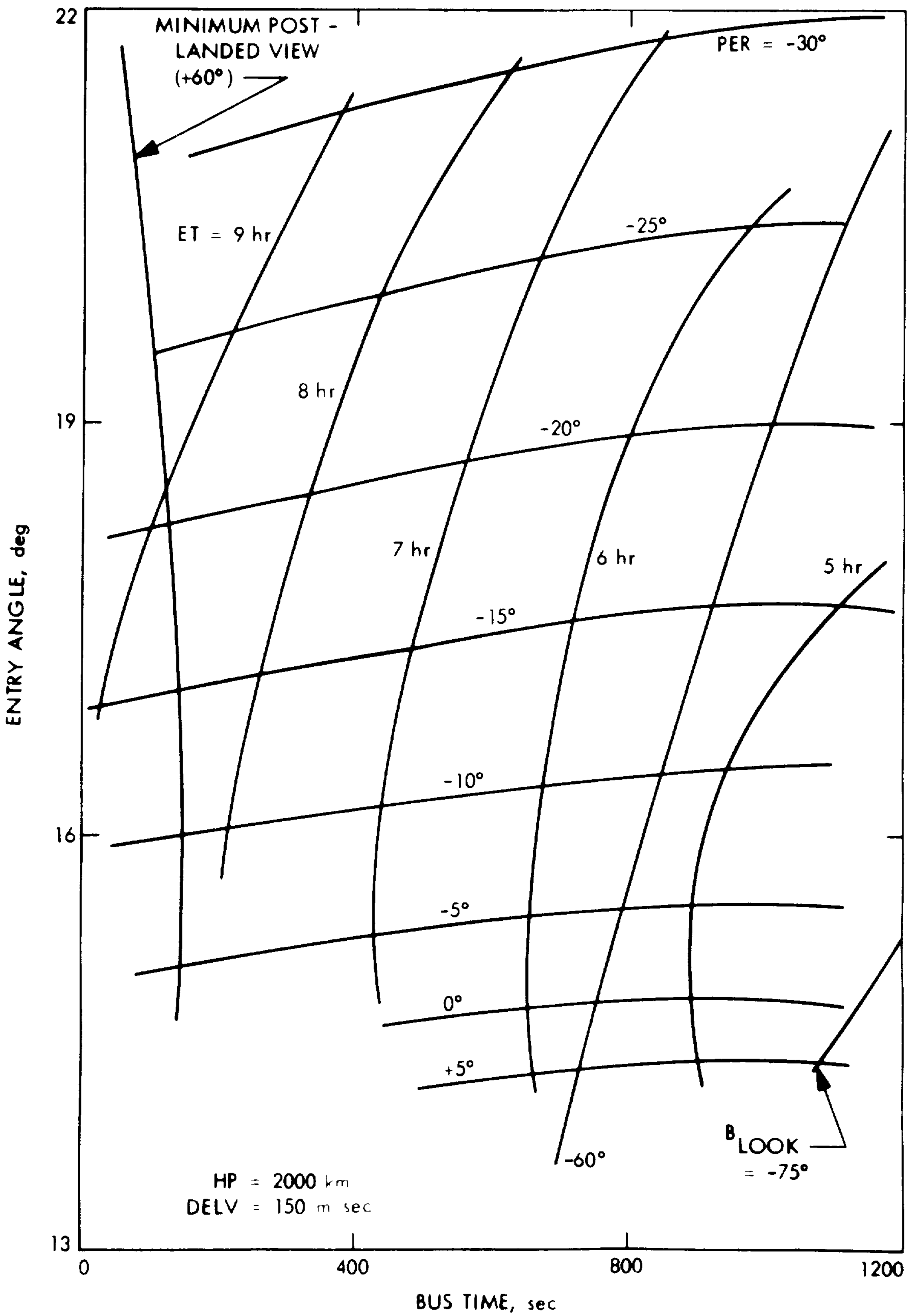

Figure 5C-12. Capsule Entry Design Chart, $h_p = 2000$ km, $\Delta V = 150$ m/sec

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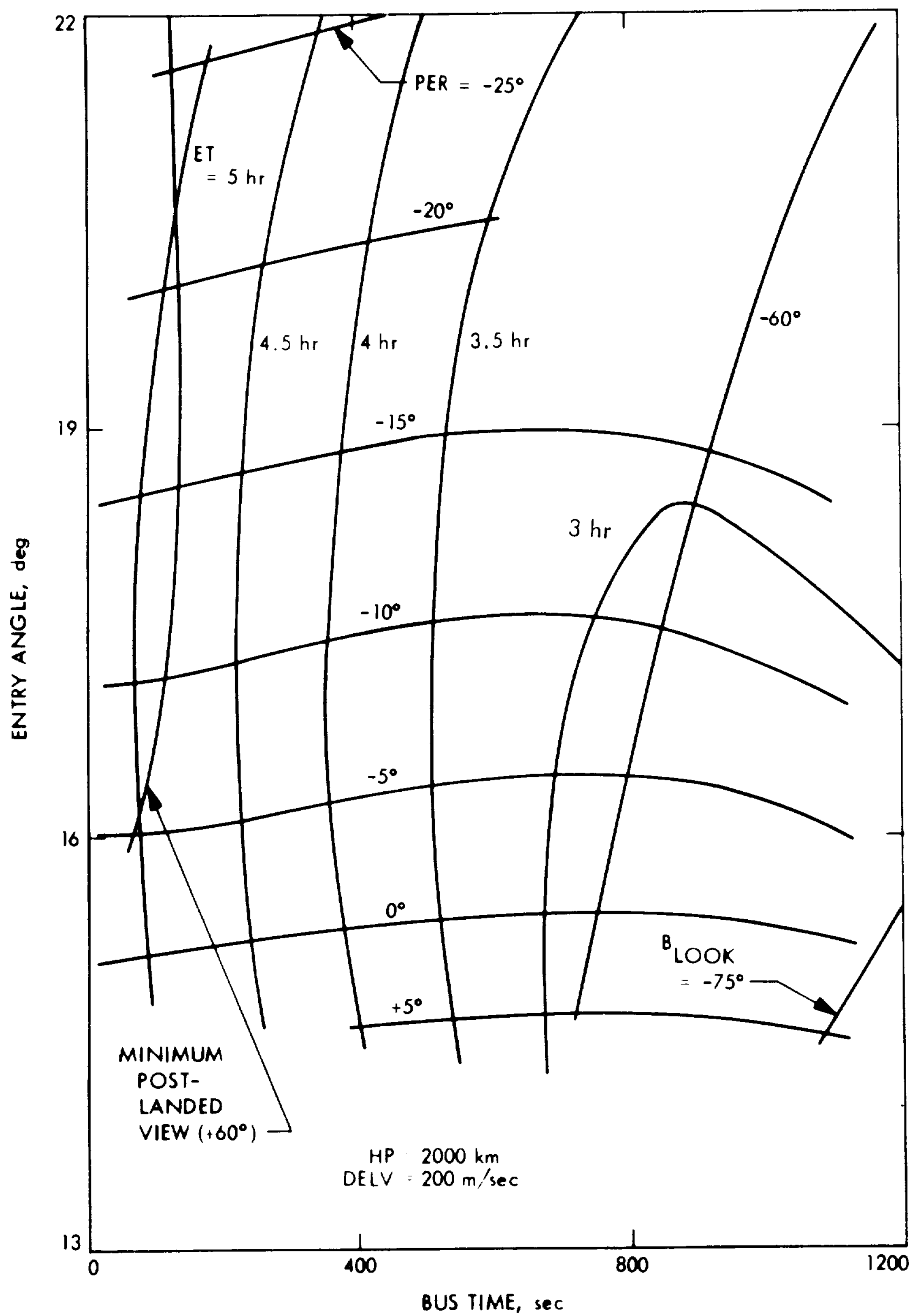

Figure 5C-13. Capsule Entry Design Chart, h_p = 2000 km, ΔV = 200 m/sec shown on these two figures for B_{LOOK} = +75 deg (negative bus time, i.e., the spacecraft would be beyond the landing site at touchdown). If entry angle is restricted to lie between 16 deg and 19 deg, PER is confined to lie between about -5 deg and -25 deg.

This information can be used with Figures 5C-14 and -15 to determine parameters regarding landing site accessibility. Figures 5C-14 and -15 show landing site latitude as a function of APS + PER for the two reference trajectory arrival dates, with h_p = 1000 km. They are based on the assumption that the spacecraft surveys the planet in nine revolutions and releases the capsule from a synchronous orbit on the tenth revolution. If the lander is to touch down in the latitude band between 0 and 30 deg North, with a solar illumination angle (SIA) approximately 15-45 deg from the evening terminator, then spacecraft orbit inclination must lie in the approximate range 20 to 60 degrees for the February 25 arrival and 25 to 60 deg for the March 20 arrival. Furthermore, using the information that $-25 \leq PER \leq -5$, it can be determined from Figure 5C-14 that, for the February 25 arrival, $-14.5 \leq APS \leq 49$, while for the March 20 arrival, $-6.5 \leq APS \leq 55.5$. Should the reader wish to determine the effects of varying entry angle and/or landing conditions, Figures 5C-9 and -11 and 5C-14 and -15 can be used to define the revised range of inclination angles and APS. It must be remembered, however, that the figures are applicable for a periapsis altitude of 1000 km and that Figures 5C-14 and -15 are for the tenth revolution after orbit insertion. Figures 5C-16 through -19 are similar to Figures 5C-14 and -15. Figures 5C-16 and -17 are for the 20th revolution after each of the two arrival dates, while Figures 5C-18 and -19 are for the 30th revolution after arrival.

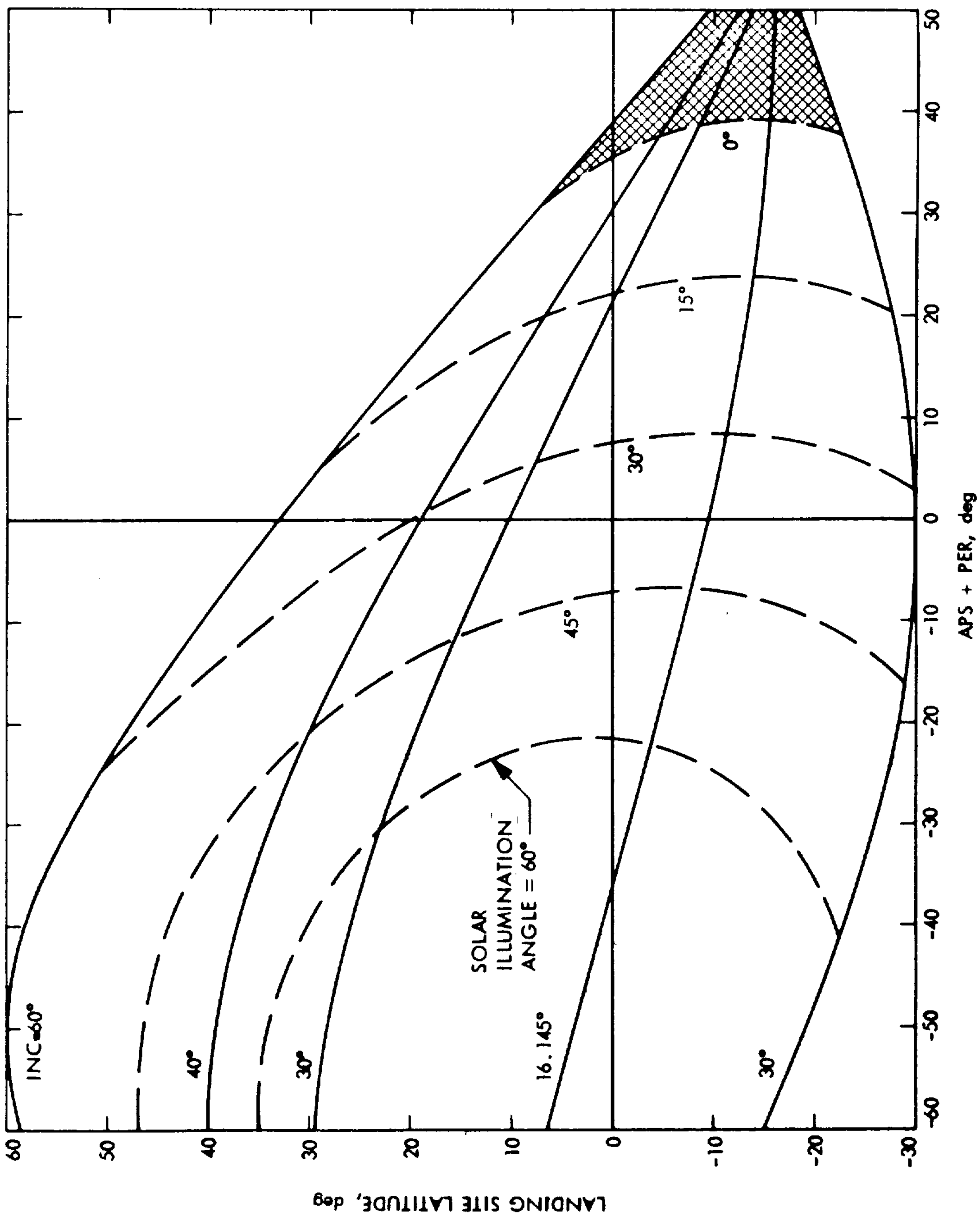

Figure 5C-14. Landing Site Accessibility, 10 days after Arrival, 2/25/74

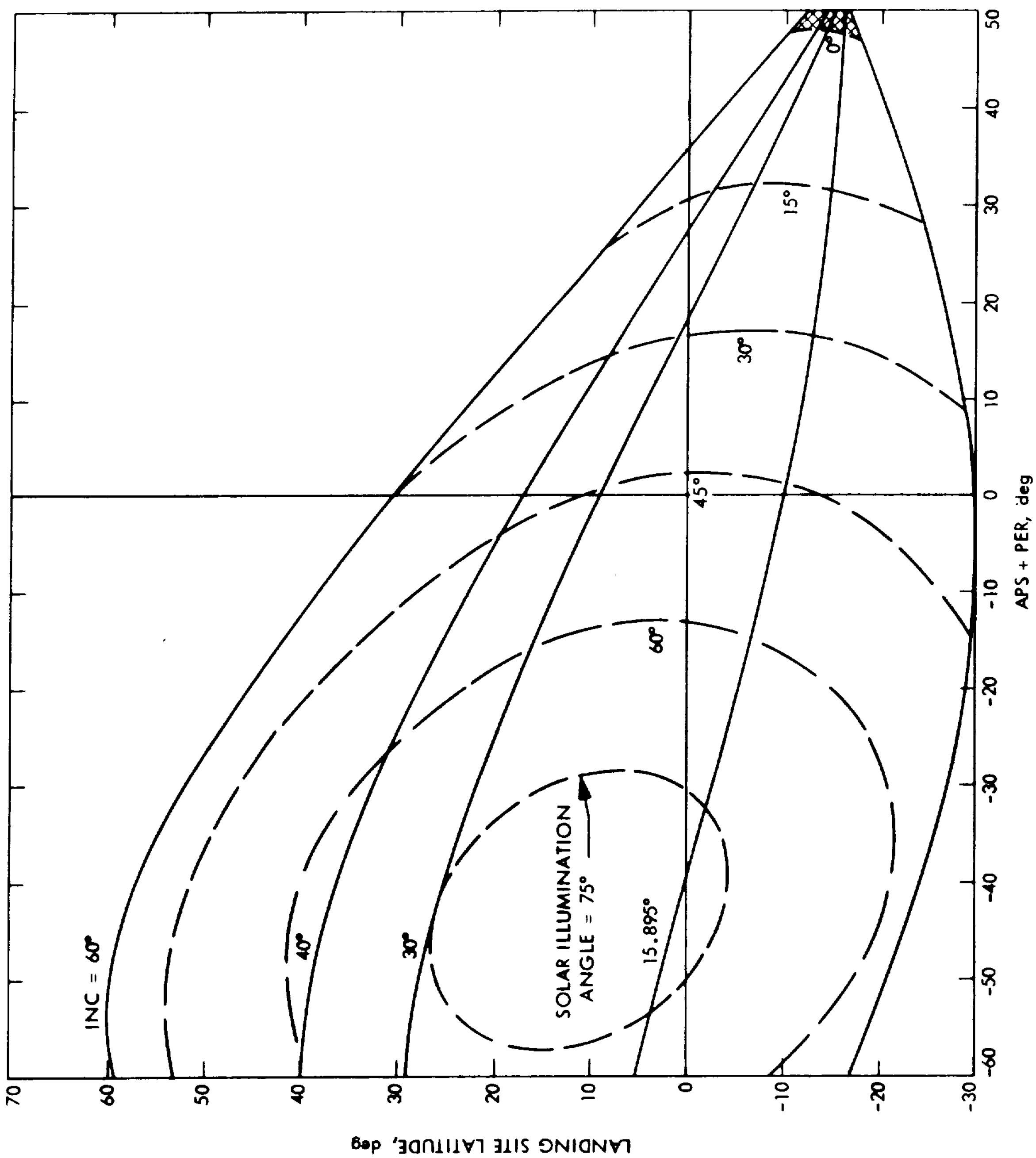

Figure 5C-15. Landing Site Accessibility, 10 days after Arrival, 3/20/74

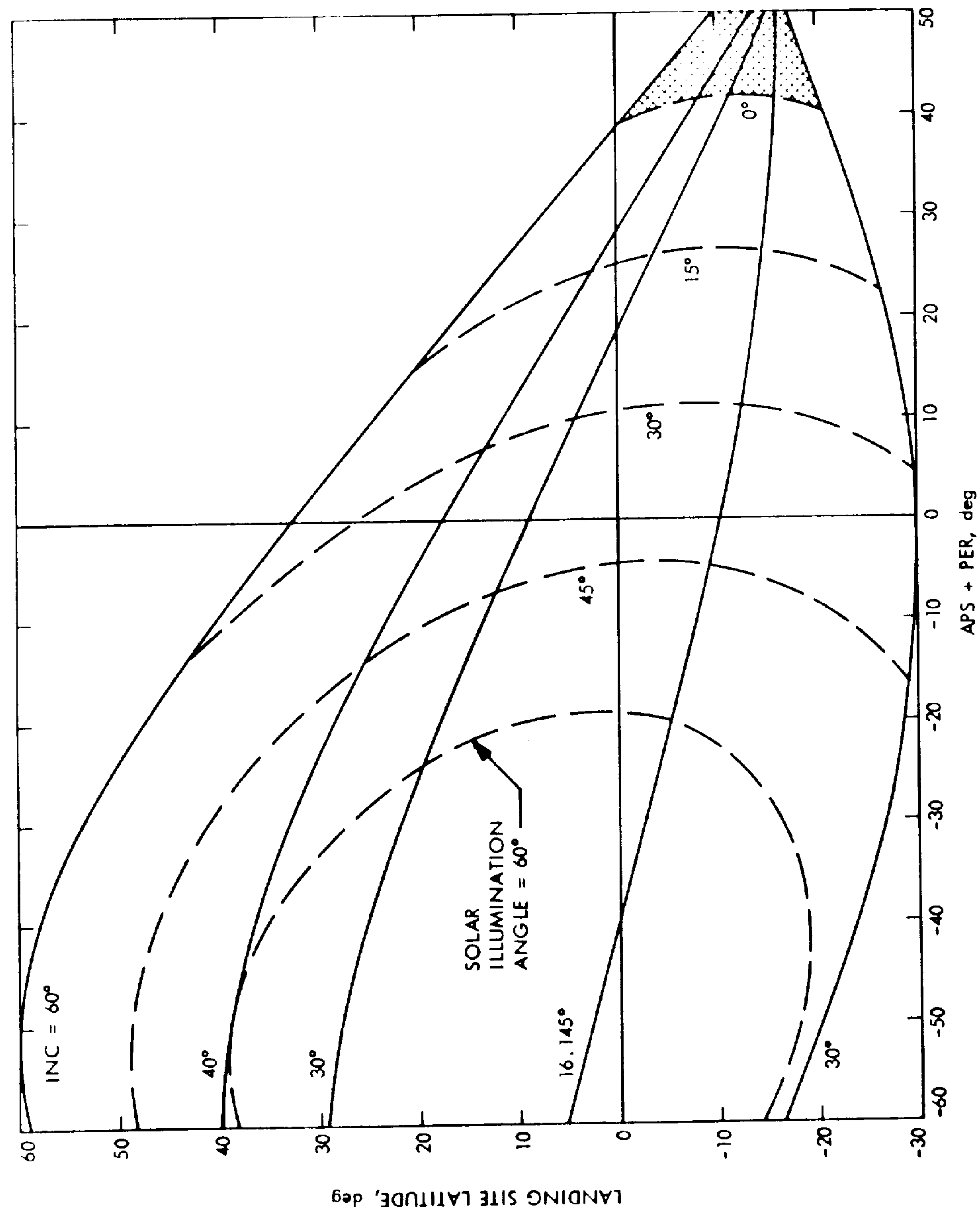

Figure 5C-16. Landing Site Accessibility, 20 Days after Arrival, 2/25/74

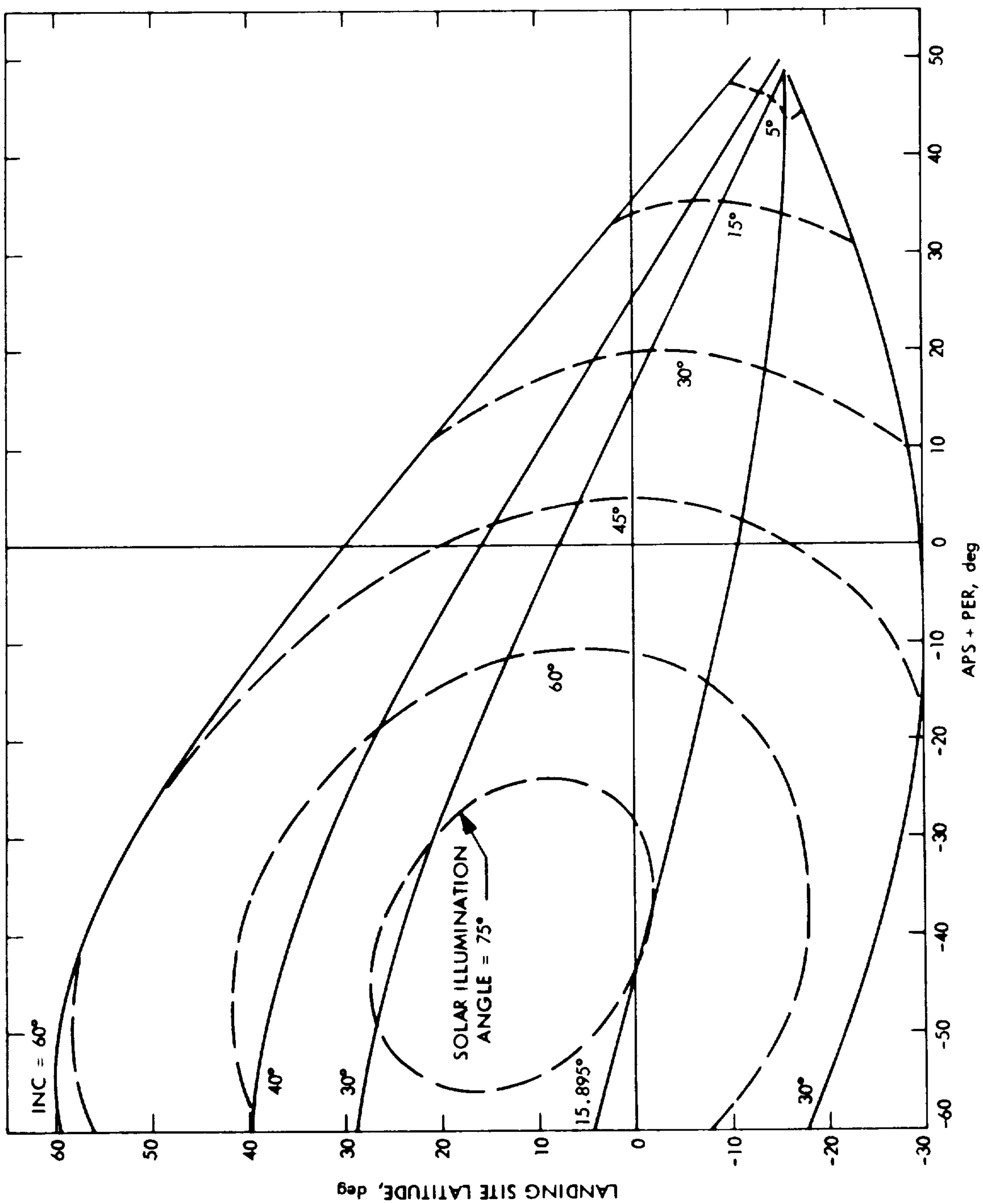

Figure 5C-17. Landing Site Accessibility, 20 Days after Arrival, 3/20/74

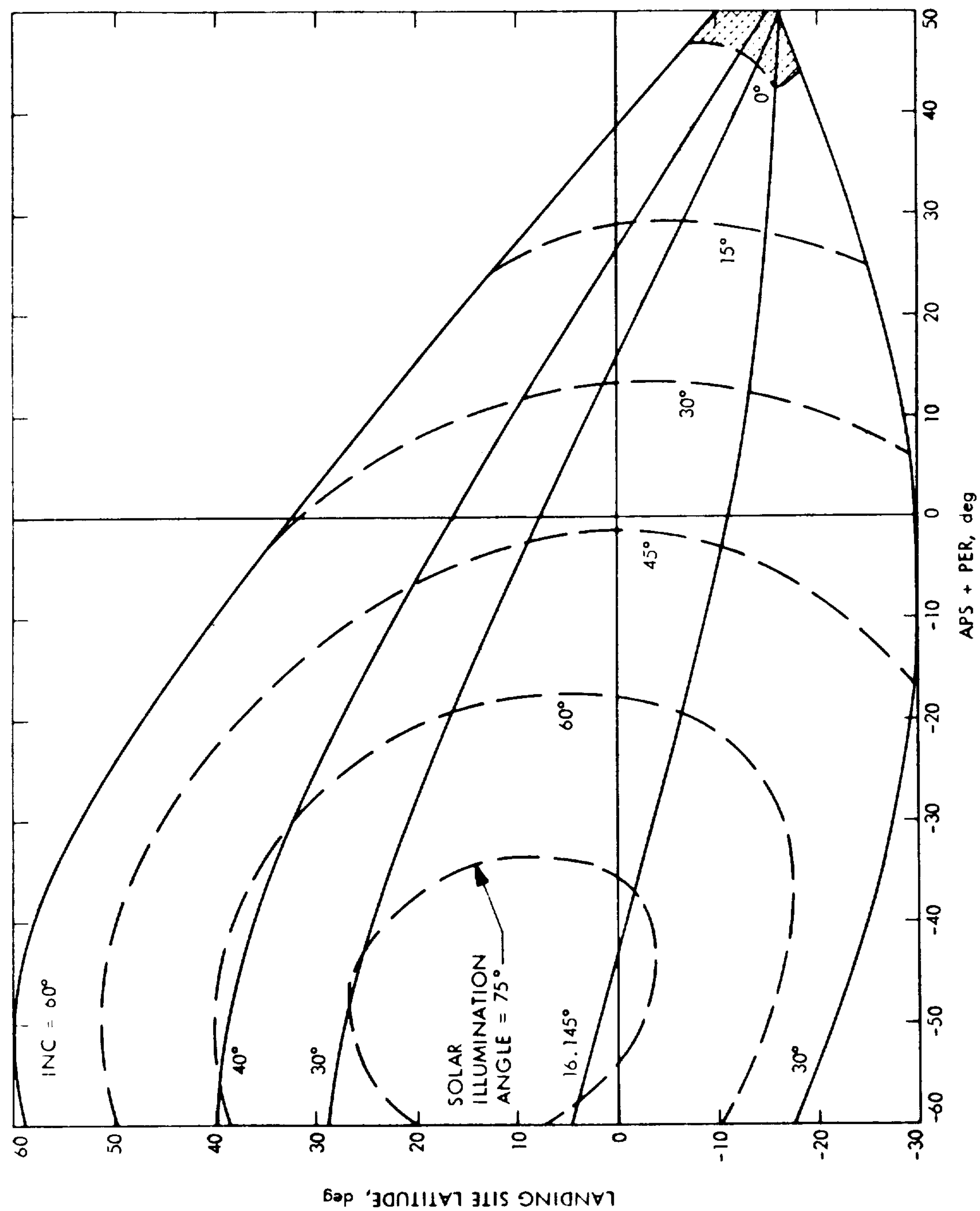

Figure 5C-18. Landing Site Accessibility, 30 Days after Arrival, 2/25/74

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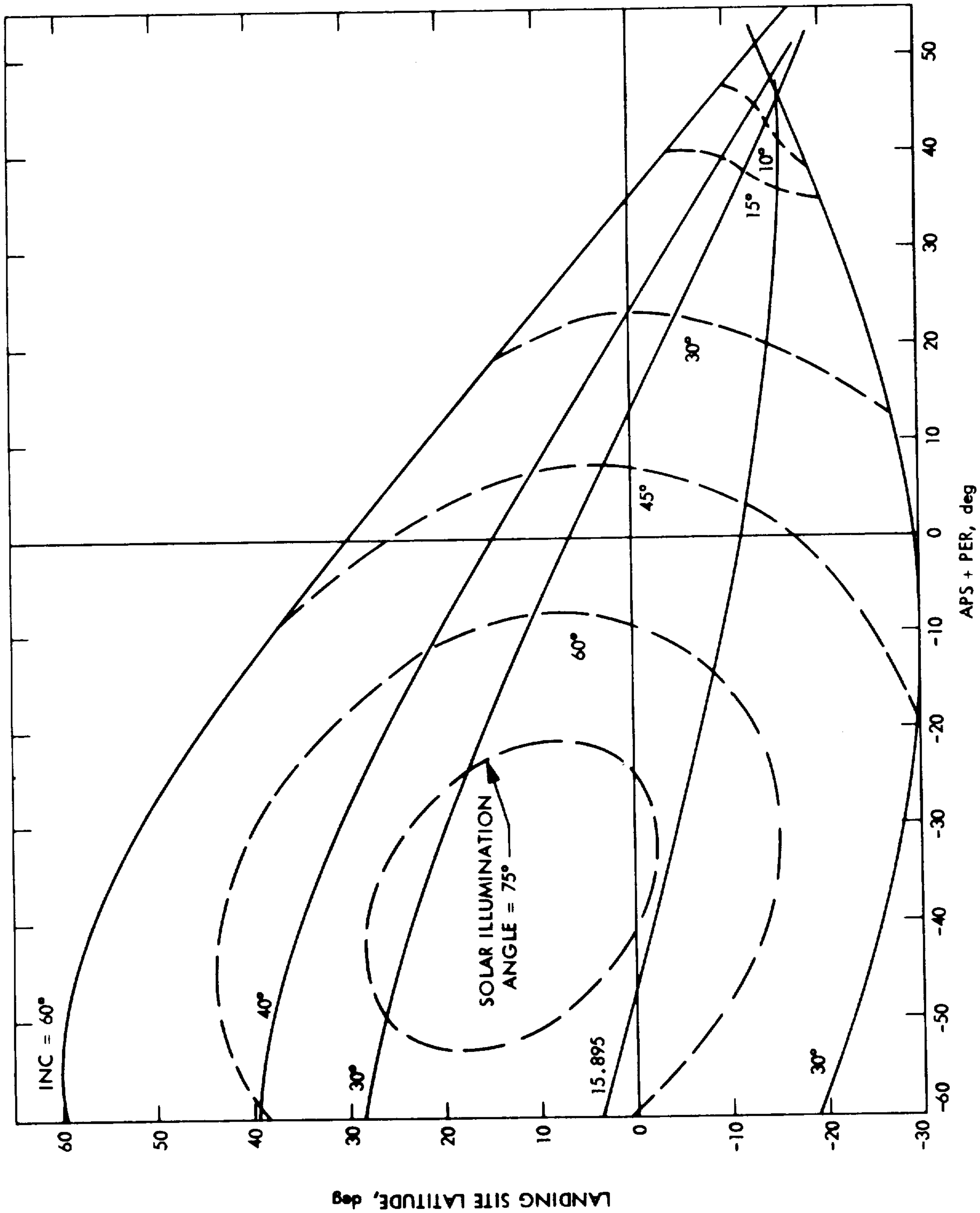

Figure 5C-19. Landing Site Accessibility, 30 Days after Arrival, 3/20/74

D. LANDING SITE SURVEILLANCE AND CHARACTERISTICS

1. Introduction

The spacecraft's first long-term function after orbit insertion is to survey the surface of Mars over all longitudes for potential landing sites. Current guidelines specify that the capsules are to land between 0 and 30 deg north latitude and that the first spacecraft is to survey potential landing sites for both landers. The ensuing paragraphs discuss the general geometry characteristics which must be considered for landing site surveillance. This is followed by a description of the planetary coverage which is obtainable from orbits which provide full longitudinal coverage in 10, 20, and 30 days for the two reference trajectories (arrival February 25 and March 20, 1974, inclination = 40 deg).

2. Viewing Geometry

As mentioned previously, surveillance of the entire latitude band between 0 and 30 degrees north is desired over all longitudes for the two orbiters.

Figure 5D-1 shows a view of the orbital ellipse for trajectory B (nominal landing latitude of 10°N) and various events concerning the surveillance. The orbiter will traverse the latitude band of 30° N to 0° under optimum lighting conditions within about 17.2 min on the first orbit. On subsequent orbits this total time will change very little. A fixed look direction of the scan platform in the plane of the orbit could be used to observe this latitude spread. However, it may be desirable to increase the scan time of this latitude spread by varying the scan platform direction in the plane of the trajectory. Thus, initially the platform could look ahead to see the latitude band and after periapsis the platform could look behind to see the latitude spread. The spacecraft could accommodate this type of surveillance by pointing the roll axis such that the cone angle of the scan direction is fixed at 90° and a clock angle variation would point the platform in the plane of the trajectory.

Figure 5D-2 shows the sub-orbiter trace across the planet for trajectory B. This figure adds dimension to Figure 5D-1. Events (1) - (4) in this figure correspond to the same events in Figure 5D-1. Prior to traversing the 30° N to 0° latitude band, such as event (1) of Figure 5D-2, the scan platform

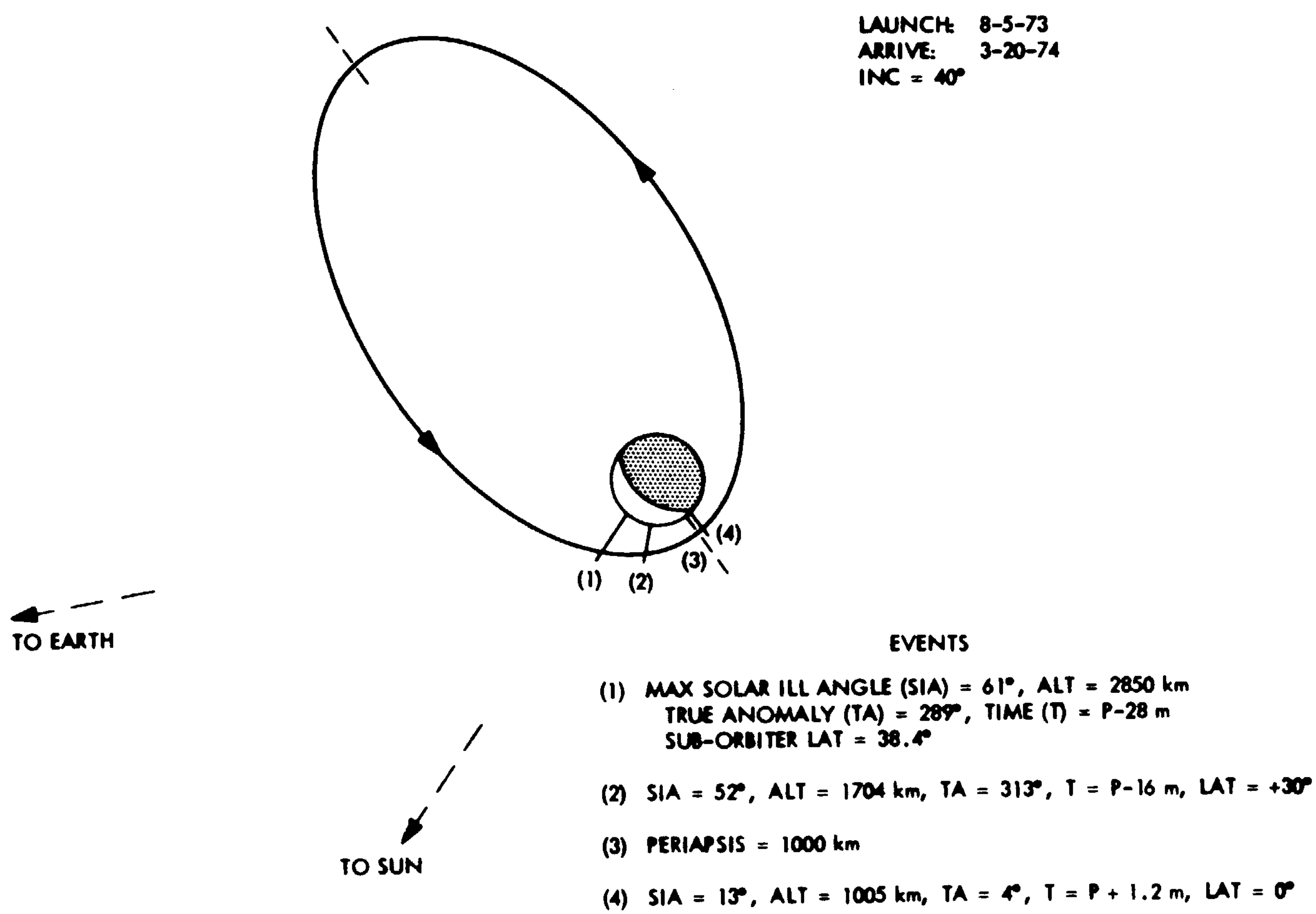

Figure 5D-1. View of Orbital Ellipse, Trajectory B

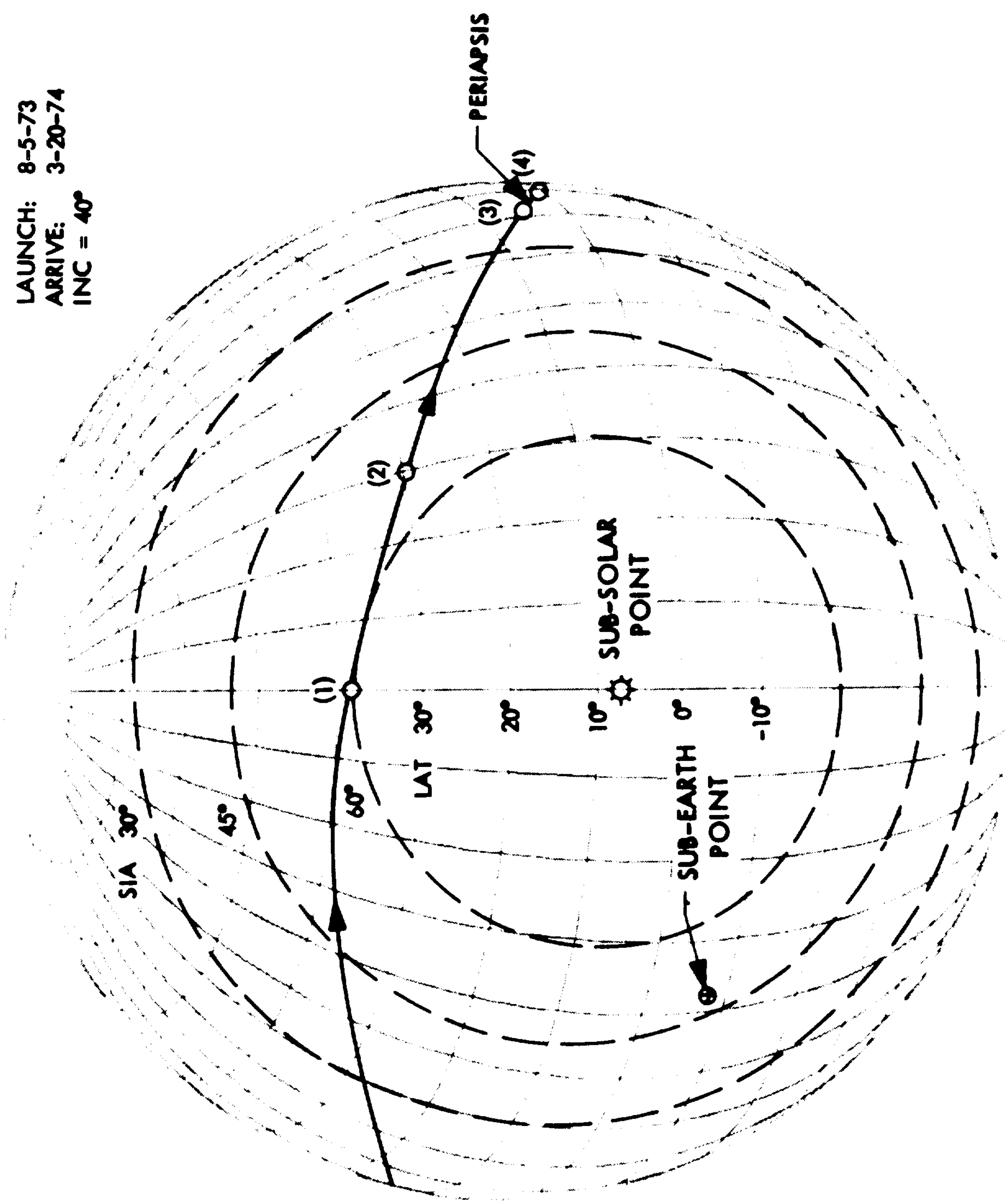

Figure 5D-2. Sub-Orbiter Trace on First Orbit, Trajectory B could be pointed down, out of the plane of the trajectory, to observe this latitude band at different longitudes and under different lighting conditions. However, the science value of this type of surveillance is not known at this time.

Figures 5D-3 and 5D-4 show plots of the latitude and altitude of the orbiter as a function of time for trajectories A and B defined in Section 5A. Also shown are various values of the solar illumination angle on the first pass. For subsequent passes the solar illumination angle will vary for a given latitude. This change will be discussed in the next paragraph. The ground speed of the orbiter while traversing the 30° N to 0° latitude band is about 3.0 km/sec. for a 1000 by 33,000 km orbit.

3. Description of Planetary Coverage

The planet is surveyed for potential landing sites over all longitudes by inserting the spacecraft into an orbit with non-synchronous period.* The period selected for landing site surveillance depends on two factors. These are: (1) the period of time which we wish to devote to full longitudinal coverage, and (2) the width of the strip of the surface viewed by the surveying instruments.

Neither of the two factors mentioned above has been spelled out definitively. However, a current ground rule is that capsule separation will occur some time between 10 and 30 days after orbit insertion. Figures 5D-5 to 5D-10 illustrate the temporal shift in longitude of the sub-spacecraft track over the planet in the latitude band 35 deg north to 10 deg south. The figures are in two sets: Figures 5D-5, -6, and -7 correspond to the shift in longitude for a spacecraft which arrives February 25, 1974 and covers the planet fully in longitude within 10, 20, and 30 days, respectively, after orbit insertion. Figures 5D-8, -9, and -10 are similar; however, they correspond to a spacecraft which

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*Figure 5C-4 illustrates the period change from synchronous to traverse all longitudes. The subject of orbital period is discussed in some detail in Section V-C, Orbit Selection and Characteristics.

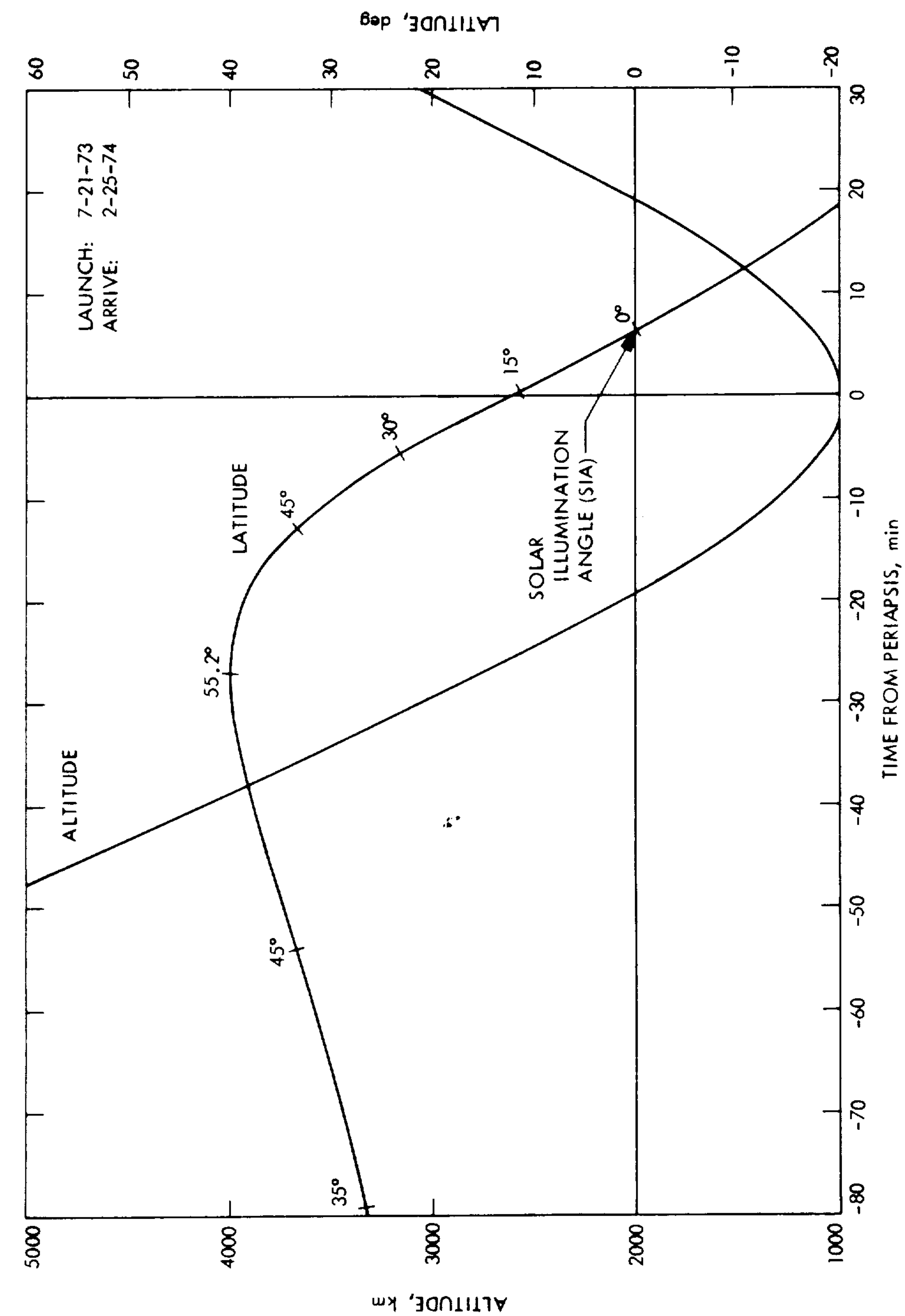

Figure 5D-3. Lattitude and Altitude of Orbiter vs. Time, Trajectory A

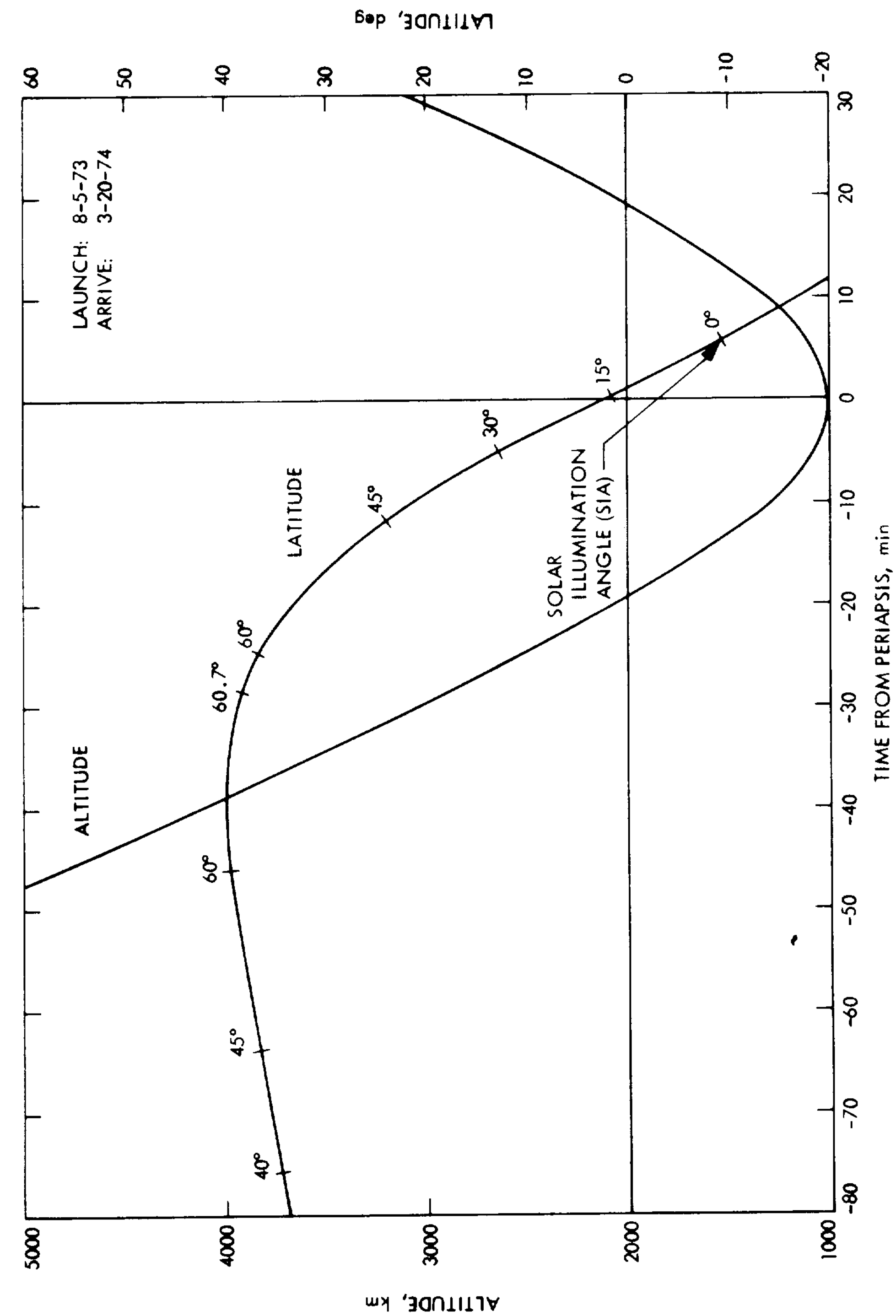

Figure 5D-4. Lattitude and Altitude of Orbiter vs. Time, Trajectory B

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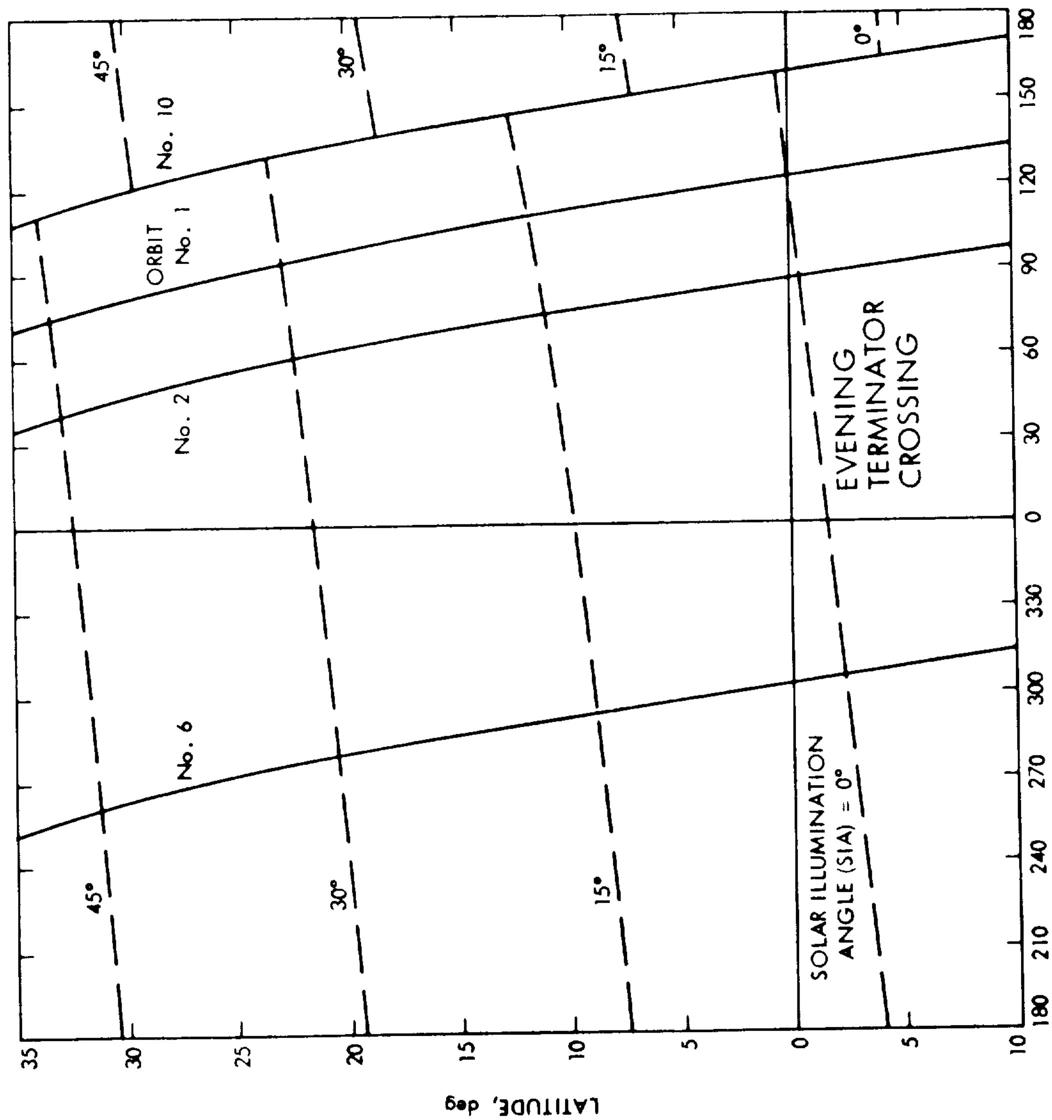

Figure 5D-5. Sub-Spacecraft Longitude Shift -- Traverse Planet in 10 Days (Arrival Date: 2/25/74)

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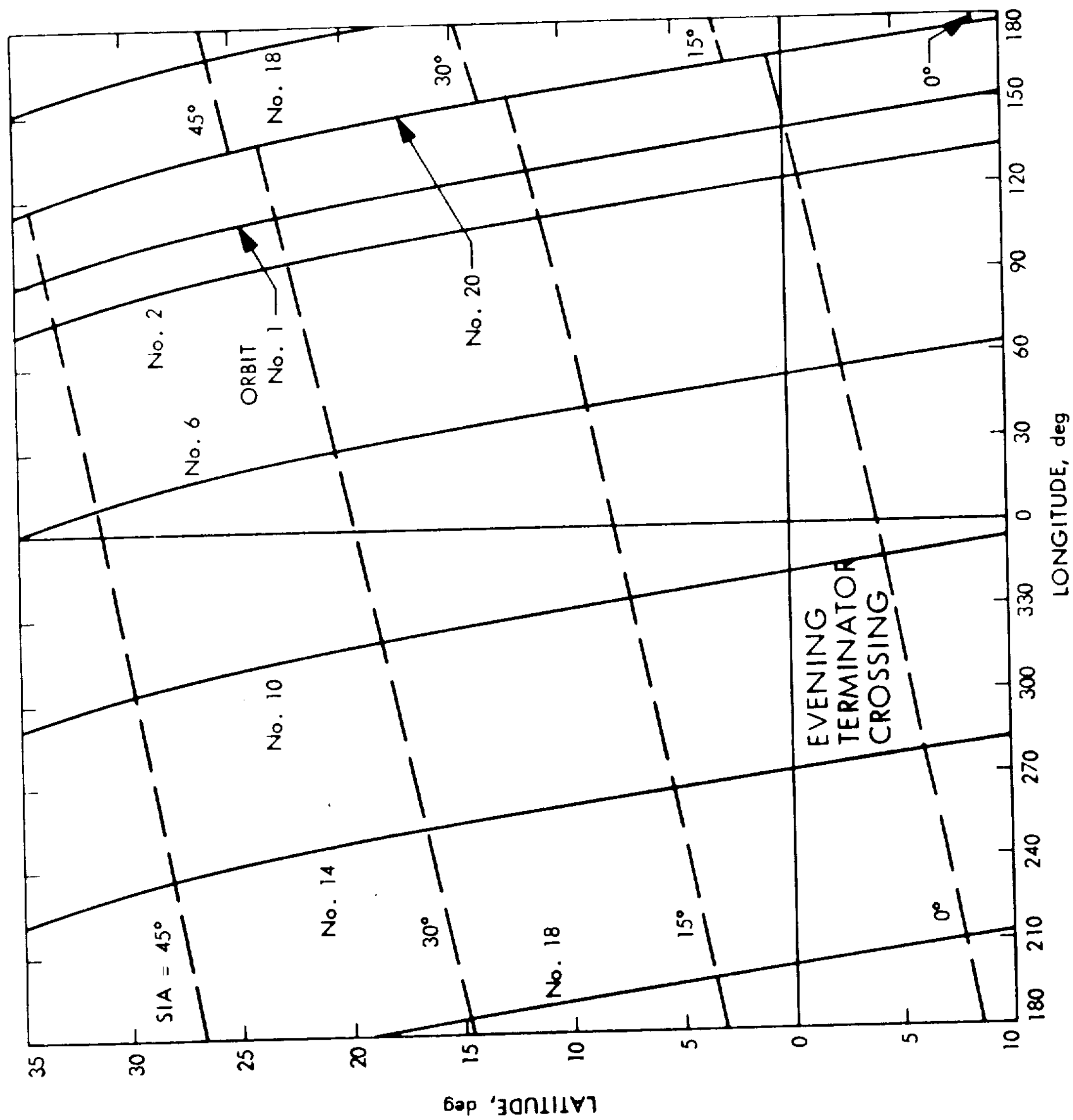

Figure 5D-6. Sub-Spacecraft Longitude Shift -- Traverse Planet in 20 Days (Arrival Date: 2/25/74)

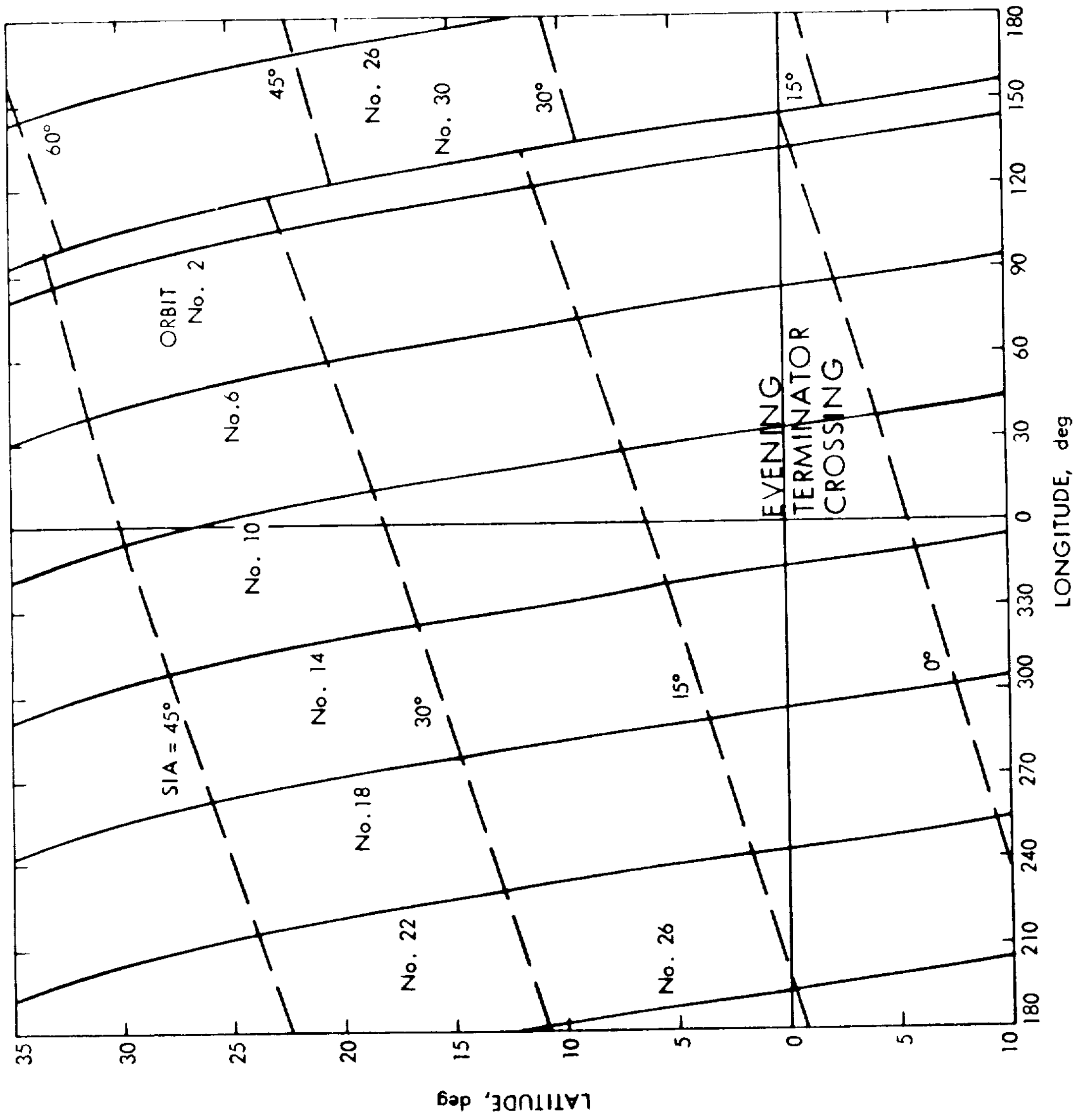

Figure 5D-7. Sub-Spacecraft Longitude Shift -- Traverse Planet in 30 Days (Arrival Date: 2/25/74)

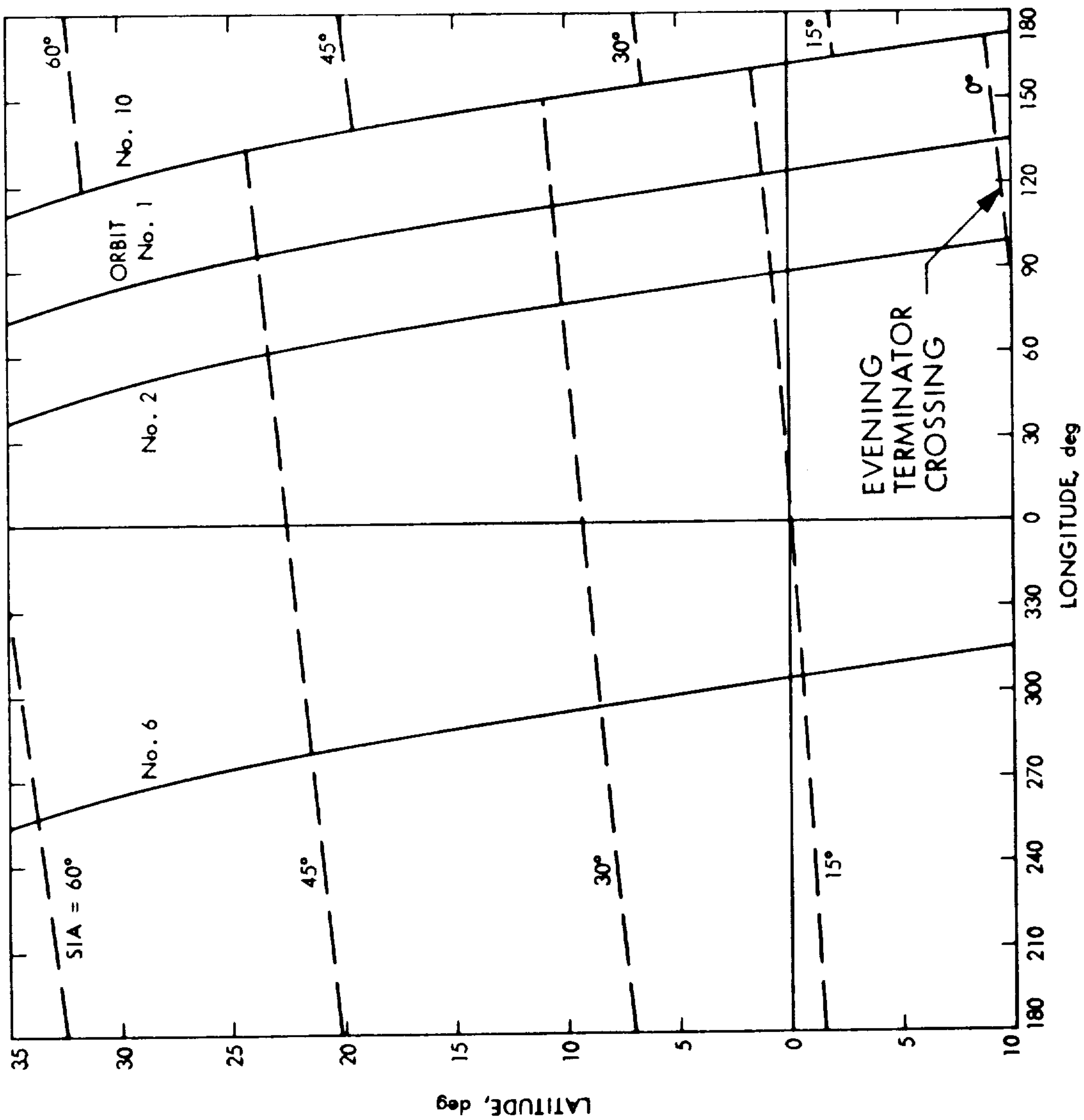

Figure 5D-8. Sub-Spacecraft Longitude Shift -- Traverse Planet in 10 Days (Arrival Date: 3/20/74)

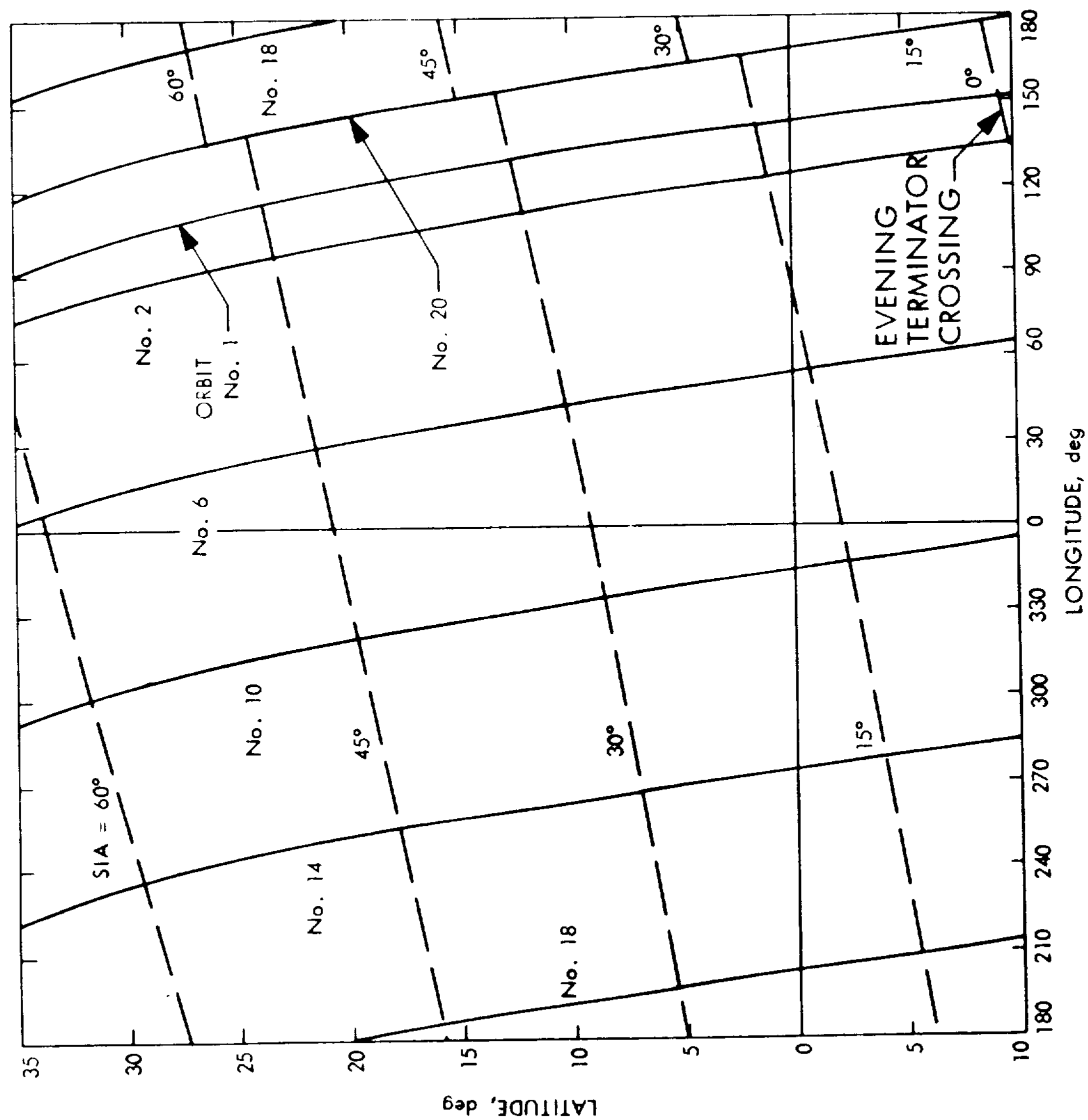

Figure 5D-9. Sub-Spacecraft Longitude Shift -- Traverse Planet in 20 Days (Arrival Date: 3/20/74)

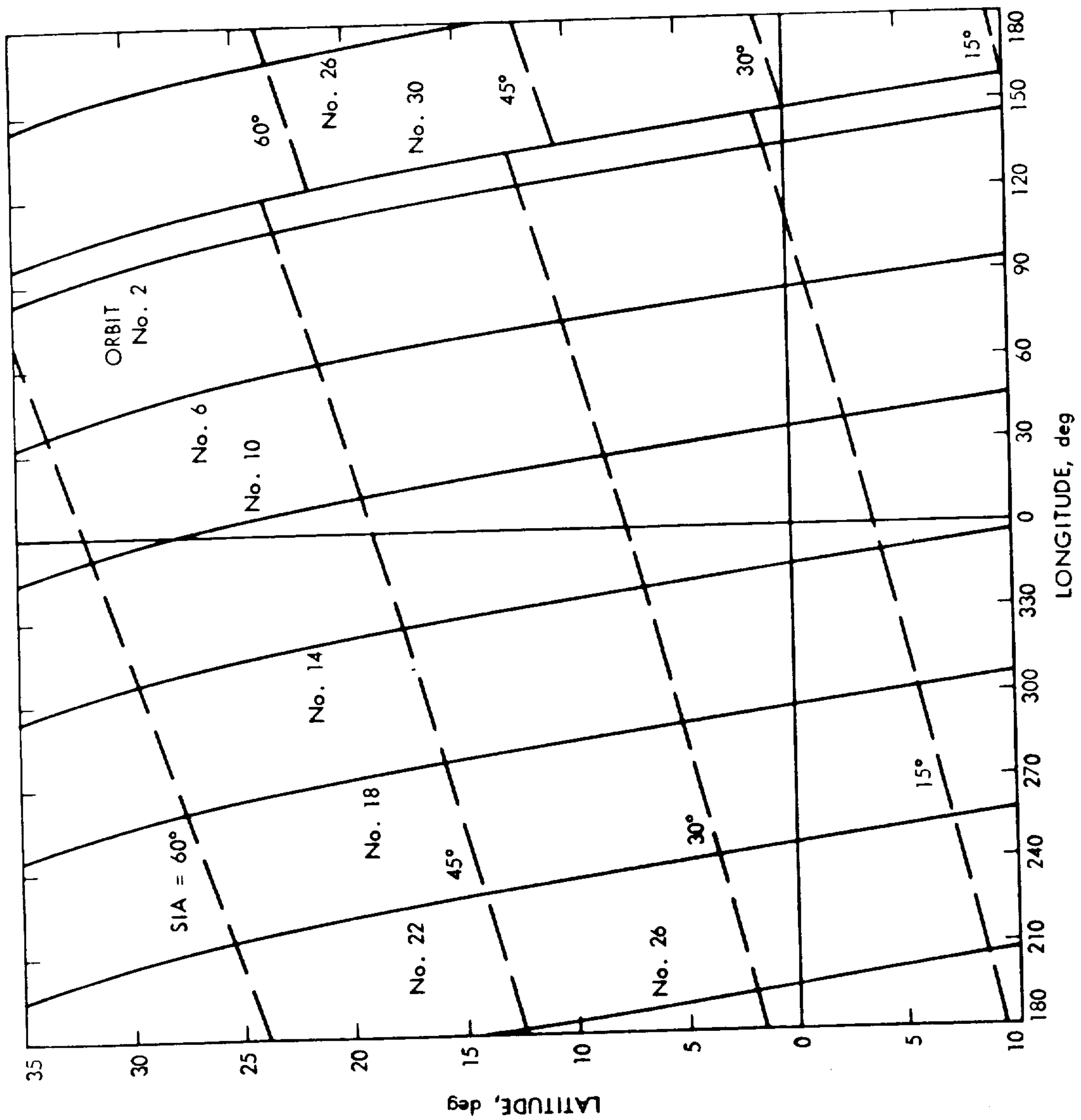

Figure 5D-10. Sub-Spacecraft Longitude Shift -- Traverse Planet in 30 Days (Arrival Date: 3/20/74)

arrives March 20, 1974.* Both sets represent an orbit inclined 40 deg. to the Martian equator with altitude at periapsis = 1000 km.

There are two sets of lines on each figure labelled, respectively, "Orbit Number" and "Solar Illumination Angle" (SIA). The line labelled "Orbit Number 1" shows the sub-spacecraft latitude as a function of longitude for the first orbital pass after orbit insertion. (The spacecraft moves from north to south, i.e., it is crossing its descending node). The sub-spacecraft track is shown on Figure 5D-5 for orbit numbers 1, 2, 6, and 10. The tracks are moving westward because the orbital period used in generating the data was chosen to be larger than synchronous. This is reasonable on the basis that such an orbit requires less insertion velocity than a sub-synchronous orbit. Figure 5D-6 shows the spacecraft track for orbit numbers 1, 2, 6, 10, 14, 18, and 20, while Figure 5D-7 shows orbits up to number 30.

The lines of constant SIA indicate lighting angles under the spacecraft. The evening terminator corresponds to SIA=0 deg. Positive values of SIA signify that the surface is illuminated, with the value of SIA corresponding to the angle of the Sun above the horizon.

Figures 5D-8, -9, and -10 correspond to 5D-5, -6, and -7, respectively but for the later arrival. Figures 5D-5 to -10 illustrate the effects of different surveillance time periods on the longitudinal spacing of the sub-spacecraft track and the variation with time in solar illumination angle over a narrow latitude band.

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*The actual sub-spacecraft longitude is a function of many variables, including the time and longitude of orbit insertion. These variables are not known explicitly, and thus neither is spacecraft longitude. Figures 5D-5 to -10 are representative of temporal shift in longitude. The reader must bear in mind, however, that the longitude on these figures is a "dummy".

E. ORBITER SPACECRAFT PROPULSION SYSTEM REQUIREMENTS AND CONSTRAINTS

1. Content

This section defines the propulsion velocity requirements of the mission.

2. Total ΔV Requirements

The total ΔV requirements are summarized below for the current conceptual spacecraft design.

1)	Midcourse maneuver	15m/sec
2)	Orbit insertion impulsive maneuver	1350
3)	Gravity burning time losses	110
4)	Pre-separation orbit trims	50
5)	Post-separation orbit trims	50

Velocity increments can be altered and interchanged in the first four ΔV allocations (1-4), as needed, on a one-to-one basis. An increase in the post-separation ΔV allocation of 50 m/sec will decrease the sum of the first four by about 30 m/sec. The orbiter propulsion system is capable of executing an unlimited number of engine firings provided propellant and pressurant remain.

a. Midcourse Maneuvers

Nominally two or three midcourse maneuvers will be performed to correct the aiming point at Mars. The time of the first maneuver is probably within launch +30 days. The second and third maneuver will probably be performed within encounter -30 days.

A total ΔV allocation of 15 m/sec (3σ) has been specified. The size of the first maneuver is conservatively estimated as 10 m/sec (3σ) and the second and third as 5 m/sec (3σ).

b. Orbit Insertion Impulse Maneuver

A total ΔV of 1350 m/sec has been allocated for the impulsive orbit insertion maneuver. This impulsive ΔV restricts the hyperbolic excess speed to be less than about 3.3 km/sec for a 1000- by 33,000-km altitude Mars orbit. For approach speeds less than 3.3 km/sec, apsidal rotations are allowed.

Figure 5E-1 shows a plot of minimum orbit insertion ΔV as a function of apsidal rotation for various approach speeds to obtain a 1000 by 33,000 km altitude elliptical orbit. The minimum orbit insertion ΔV corresponds to the optimum impulsive transfer for which the approach hyperbola is nearly tangential to the desired ellipse. However, for finite apsidal rotations the optimum impulsive transfers occur for the hyperbolas which pass slightly inside the ellipse, at one of the two intersections. This optimum bias is usually between 0 to 50 km for apsidal rotations less than ±30° (see Figure 5E-2).

Since the final hyperbolic periapsis cannot be controlled precisely due to maneuver execution errors and orbit determination uncertainties, it may be desirable to bias the nominal hyperbola inward to insure that the final flyby hyperbola will intersect the desired ellipse. However, an additional inward bias will cause the orbit insertion ΔV to increase. Figure 5E-2 shows a plot of impulsive orbit insertion ΔV for inward biases from the tangential transfers. Note that the minimum impulsive transfer already possesses a small inward bias. Figure 5E-3 shows a graphical explanation of the bias.

The philosophy of intentional biasing of the nominal hyperbola to insure intersection with the desired ellipse for dispersed trajectories, must be examined in more detail. An alternate policy would be to accept a different apsidal rotation of the ellipse if the hyperbolic periapsis is high or to accept a higher elliptical periapsis with the correct apsidal rotation and then to trim the periapsis while in orbit.

c. Gravity Burning Time Losses

A ΔV allocation of 110 m/sec has been tentatively assigned for the orbit insertion gravity burning time losses. This ΔV is the difference

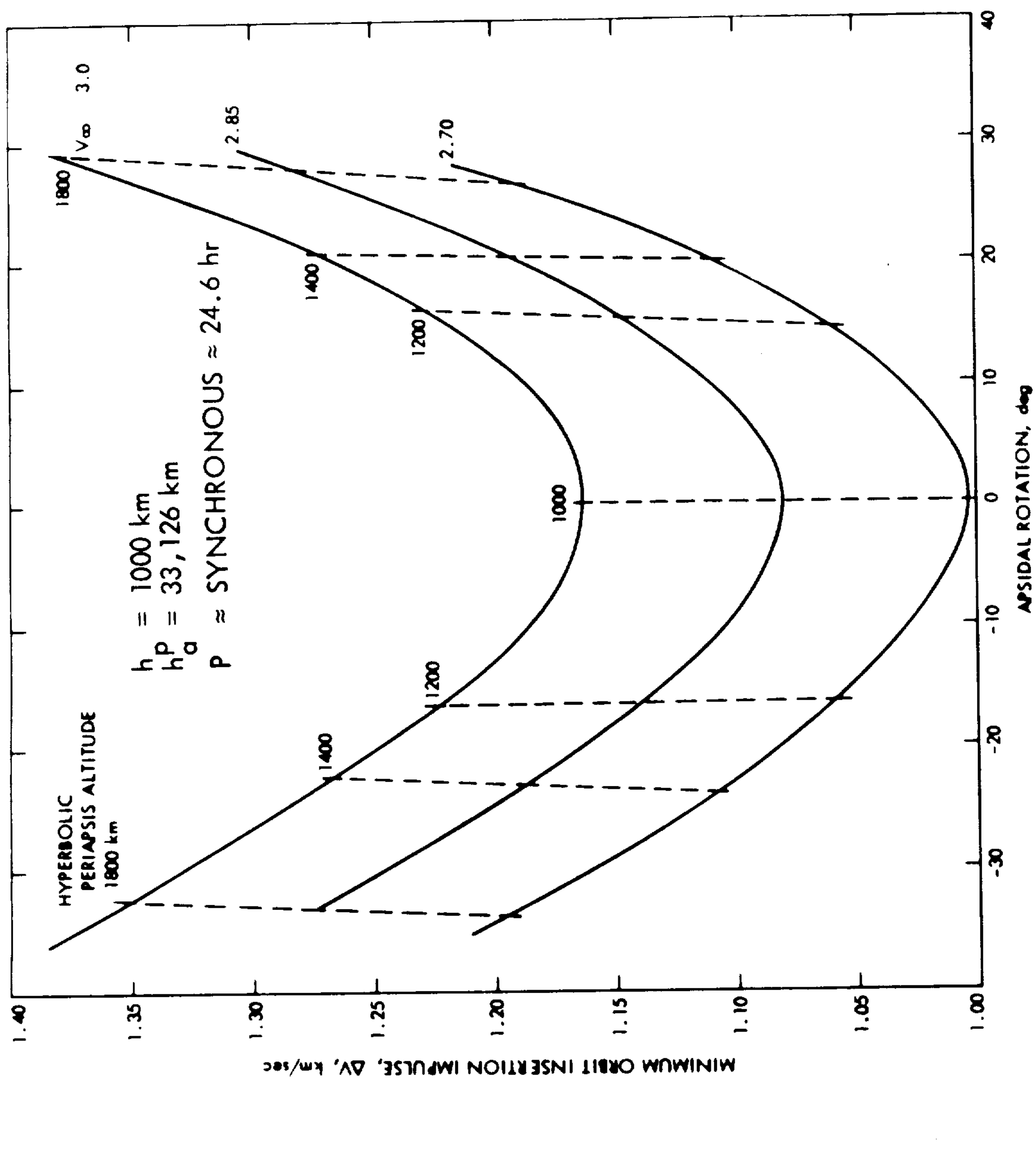

Figure 5E-1. Minimum Orbit Insertion Impulse for Various Approach Speeds and Apsidal Rotations

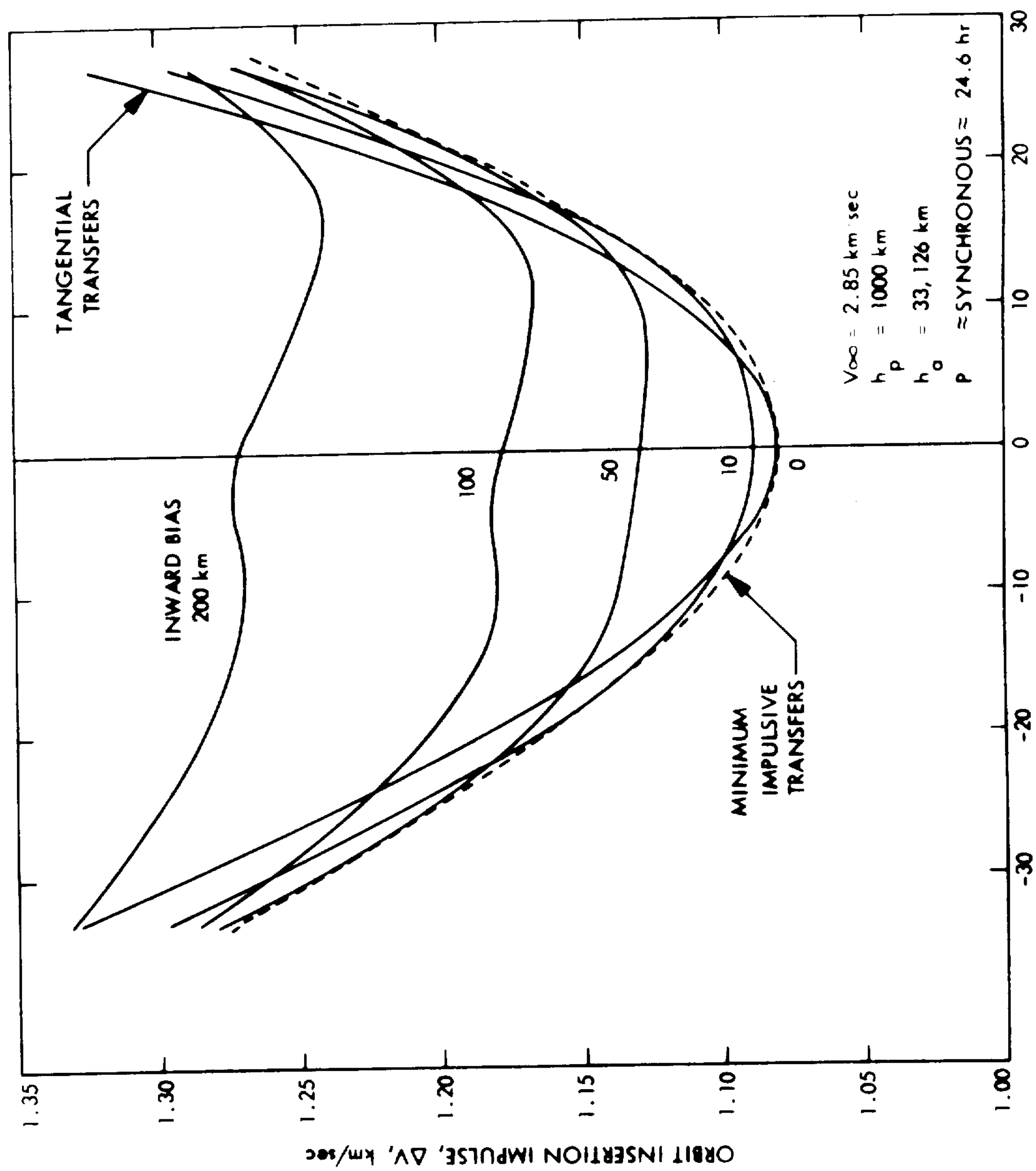

Figure 5E-2. Orbit Insertion Impulse for Intentional Biases with Apsidal Rotation

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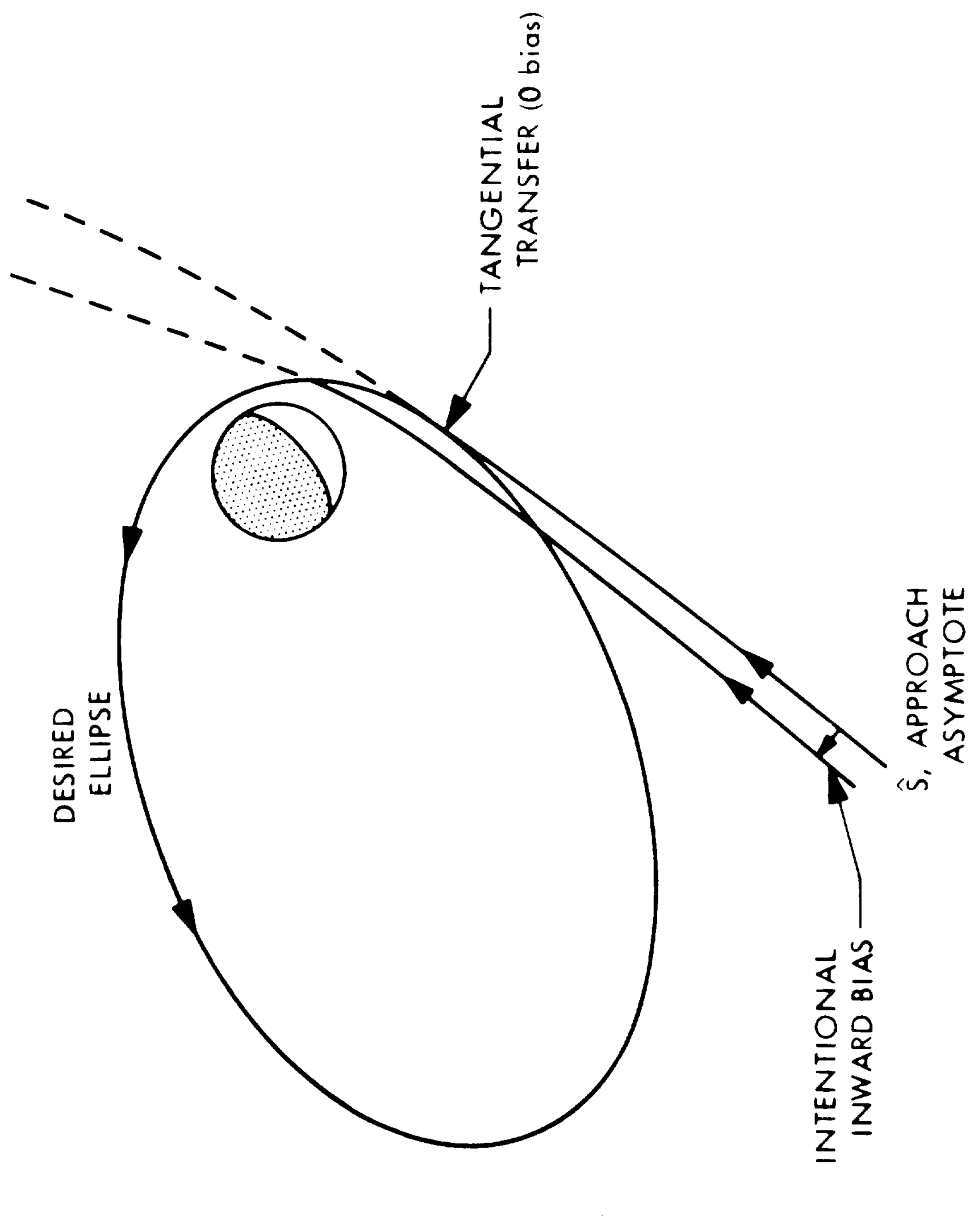

Figure 5E-3. Definition of Intentional Bias between the impulsive orbit insertion ΔV and the finite burn ΔV and is thus the gravity burn loss.

Figure 5E-4 shows a plot of gravity losses for fixed pointing of the orbit insertion thrust as a function of the initial thrust to weight ratio (T/W). For a T/W of .041, corresponding to the current spacecraft and propulsion system design, the 110 m/sec allocation will restrict the approach speeds to be less than about 3.0 km/sec. Additional ΔV losses will have to be allocated if higher approach speeds are used. If a gravity turn or pitch turn insertion maneuver is used, the gravity losses may be reduced by as much as 50%.

The gravity losses of Figure 5E-4 correspond to an apsidal rotation of zero degrees. The gravity losses will generally be less for apsidal rotations resulting from minimum impulsive transfers. Figure 5E-5 shows the gravity losses for various apsidal roations for T/W=.05 and V_∞ = 2.85 km/sec. The dashed line corresponds to the minimum impulsive transfers. Also shown are the gravity losses for intentional biasing of the hyperbola, corresponding to Figure 5E-2.

d. Pre-Separation Orbit Trim

A total pre-separation orbit trim of 50 m/sec has been tentatively allocated. One or more orbit trims would be used to adjust the periapsis altitude if necessary, to adjust the orbital period for optimum landing site surveillance if necessary, and to synchronize the orbital period with the desired landing site once the surveillance is completed.

Figure 5E-6 shows the orbit trim requirements to change the orbital period, but to keep the same periapsis altitude of 1000 km. Figure 5E-7 shows the orbit trim requirements to change the period and to also change the periapsis altitude at 3 hours before periapsis passage (P-3hr). Figure 5E-8 defines the thrust application angle of Figure 5E-7.

The synchronous orbit trim maneuver will probably be made near periapsis to minimize the ΔV requirements. It may be desirable to perform the maneuver at an hour or so away from periasis if the orbiter science data is still desired at periapsis passage. However, changing the orbital

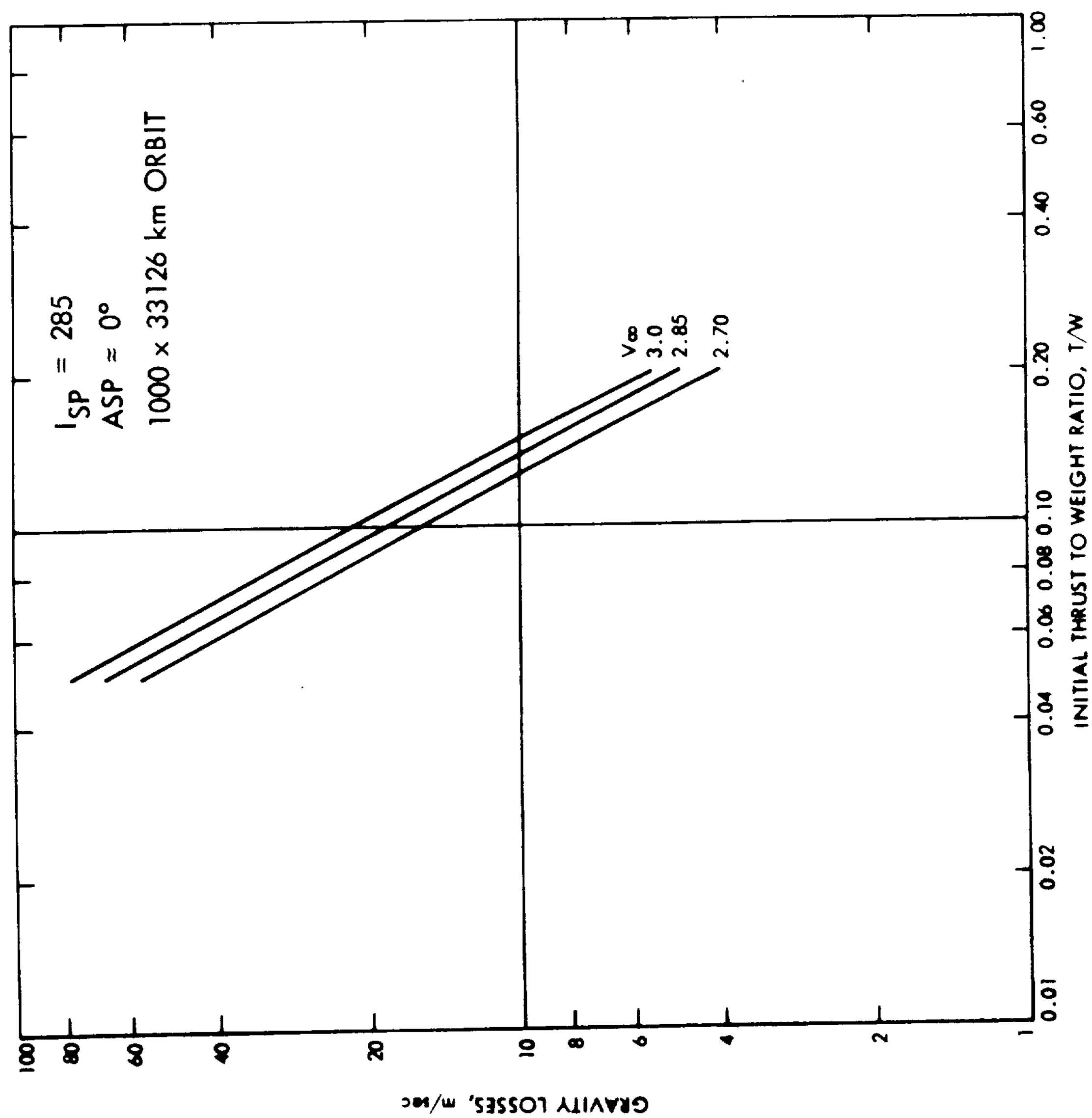

Figure 5E-4. Orbit Insertion Gravity Losses for Fixed Direction of Thrust Vector

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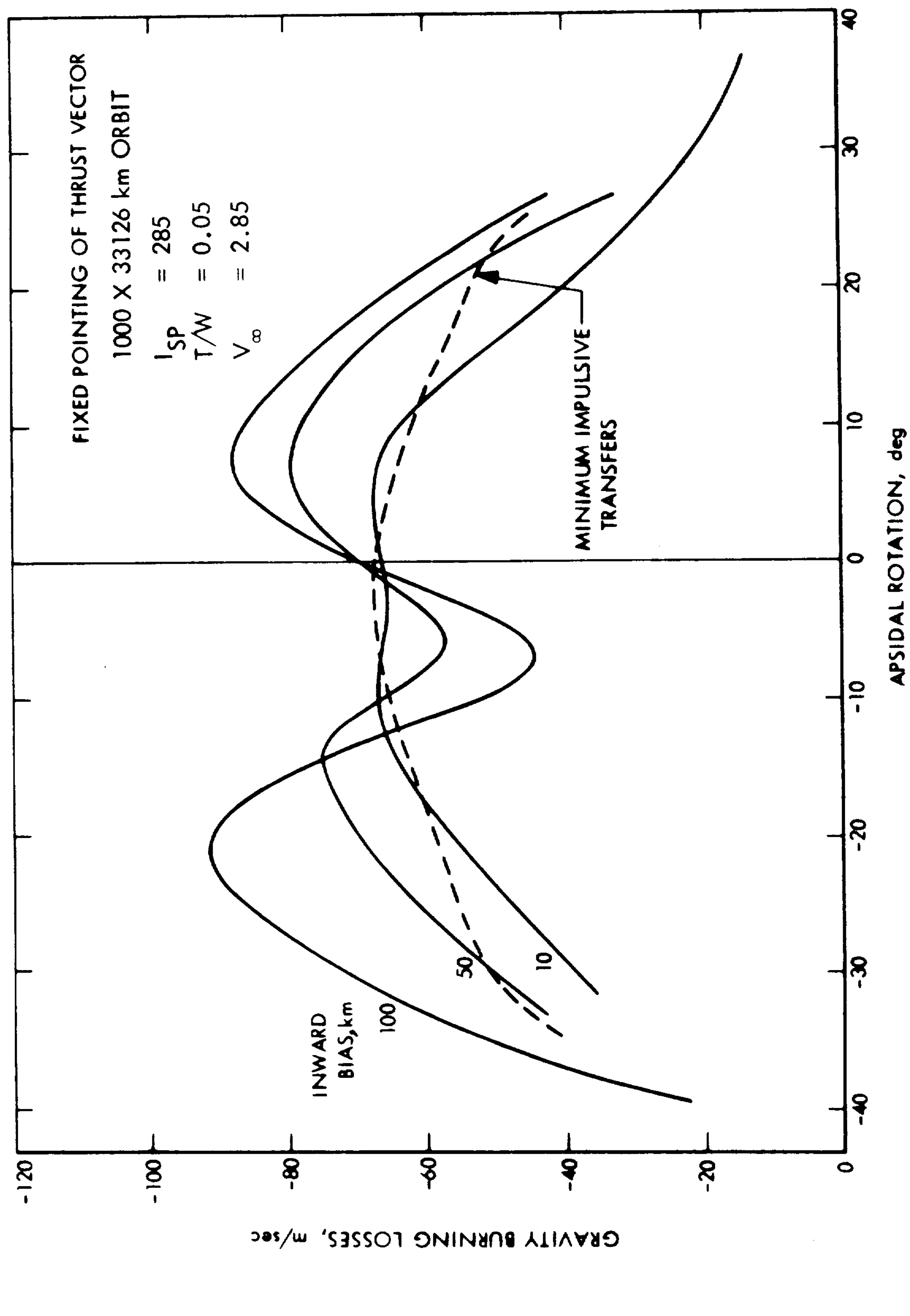

Figure 5E-5. Orbit Insertion Gravity Losses for Intentional Biasing

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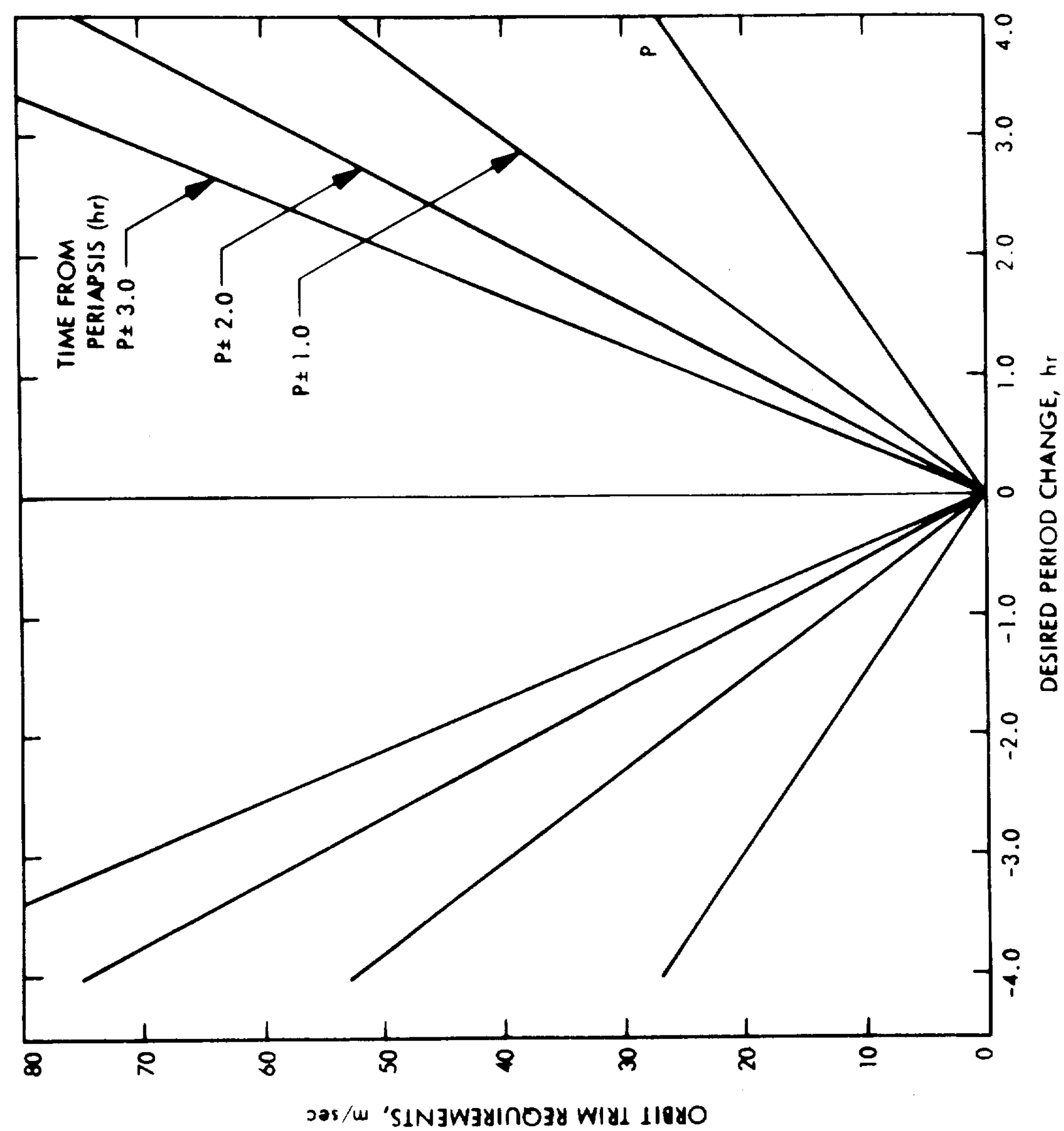

Figure 5E-6. Orbit Trim to Change Period

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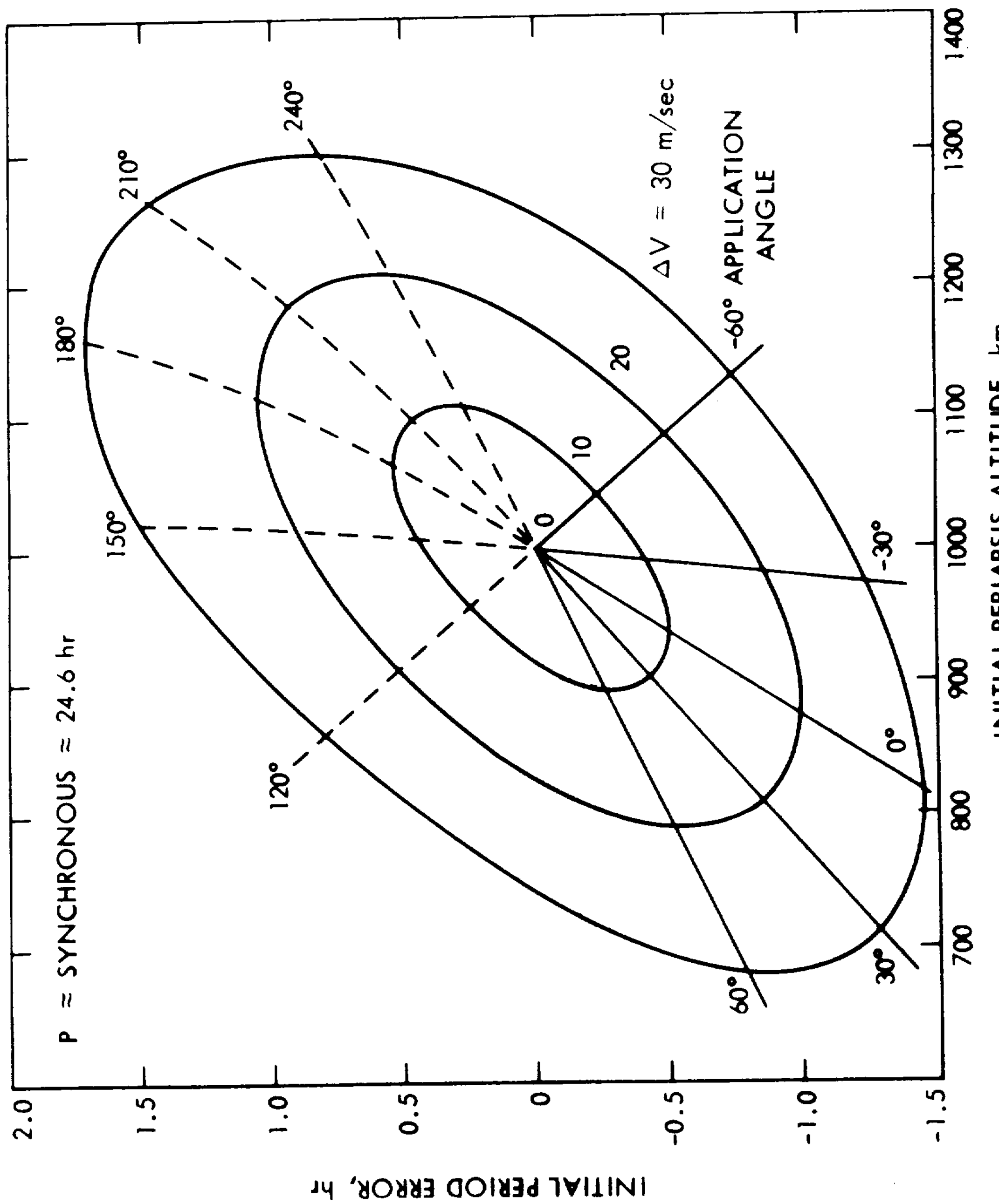

Figure 5E-7. Orbit Trim at P-3 hr to Establish a Periapsis Altitude of 1000 km and to Change Period

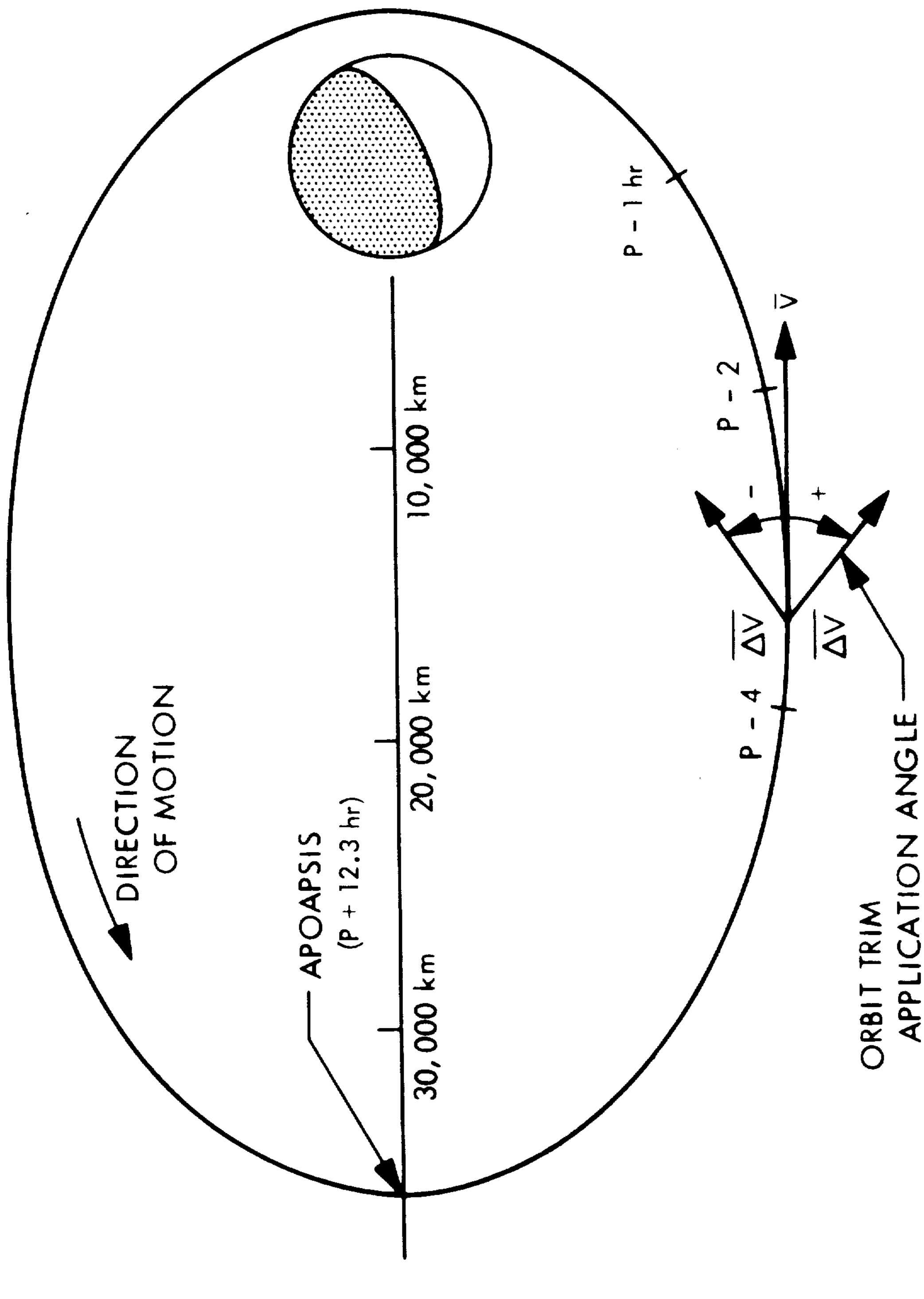

Figure 5E-8. Definition of Orbit Trim Application Angle period by about 2.5, 1.2, and 0.8 hours at periapsis will require about a 16, 8, and 5 m/sec orbit trim maneuver respectively (see Figure 5E-6). These period changes would be required to synchronize initial surveillance orbits for which the consecutive sub-orbiter traces on Mars were separated by about 36°, 18°, and 12° respectively.

e. Post Separation Orbit Trim

A total of 50 m/sec has also been tentatively allocated for the post separation maneuver. The first trim would be used to de-synchronize the orbit after the few-day relay link communications is completed. This would be done to allow TV mapping over all longitudes. Figure 5E-6 can be used to find the allowable period change for various orbit trim maneuvers and times, assuming the 1000 km periapsis altitude is kept unchanged.

The de-synchronizing maneuver will probably be made near periapsis to minimize the ΔV requirements. Changing the period at periapsis by about 0.8 hrs will allow the orbiter to pass over the lander every 30 days and will require a ΔV of about 5 m/sec.

Additional back-up maneuvers may be needed as follows: to re-synchronize the orbit over the lander on a subsequent pass in case the lander direct link communication fails; and to provide a back-up planetary quarantine maneuver in case the orbit lifetime is not adequate.

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F. LAUNCH VEHICLE SYSTEM

1. Configuration

The Viking mission is predicated on the availability of two (2) Titan IIID/Improved Centaur launch vehicles as required to launch two (2) spacecraft during the 1973 Mars opportunity. The vehicle stack-up is shown in Figure 5F-1. The assumed Centaur fairing is shown in Figure 5F-2.

The Titan consists of two solid rocket motors of five segments each, and two liquid stages substantially as configured for the Titan IIIC less Transtage. The two engines of the first liquid stage have nozzle expansion ratios of 15:1 rather than 12:1 as in the current vehicle. The Titan guidance system is eliminated, as guidance and timing functions are provided by the Centaur. The Titan autopilot, of course, is retained. Structural changes to accommodate theCentaur will be made primarily at theTitan/Centaur interface.

The improved Centaur is currently being defined, but is expected to have the nose fairing structurally attached to the Centaur insulation panels at a field joint. The total assembly would be jettisoned in radial segments. If this is done during the burn of the last stage of Titan, the potential contamination of the orbiter from the Titan separation rockets will require investigation at an early date. Jettison during the Centaur phase would eliminate this concern, but at a cost in performance. The Improved Centaur has other new features, including an extension of the allowable parking orbit coast period to one hour.

As discussed later, the nose fairing is assumed to accommodate the total encapsulation of the spacecraft in a clean room prior to mating the encapsulated assembly to the launch vehicle.

2. Launch Complex and AGE

A single launch pad (41) is expected to be available in 1973 for all Titan/Centaur launches. This assumption dictates an available launch period of sufficient duration to launch both Viking spacecraft with due allowance for (1) possible pad damage, (2) possible curtailment in pad availability time due to USAF programs at ETR, and (3) checkout and Combined Systems Testing of

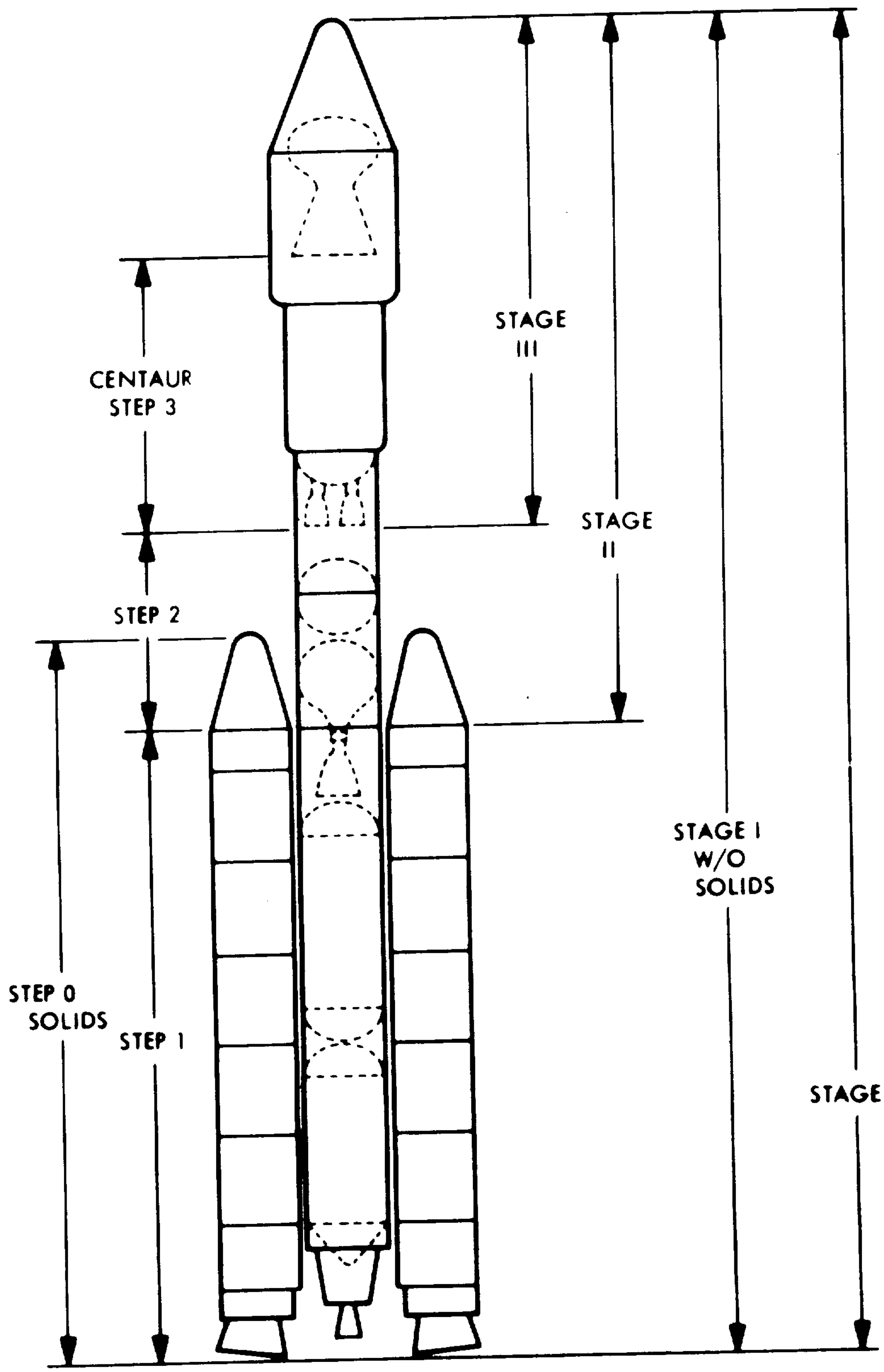

Figure 5F-1. Titan Centaur

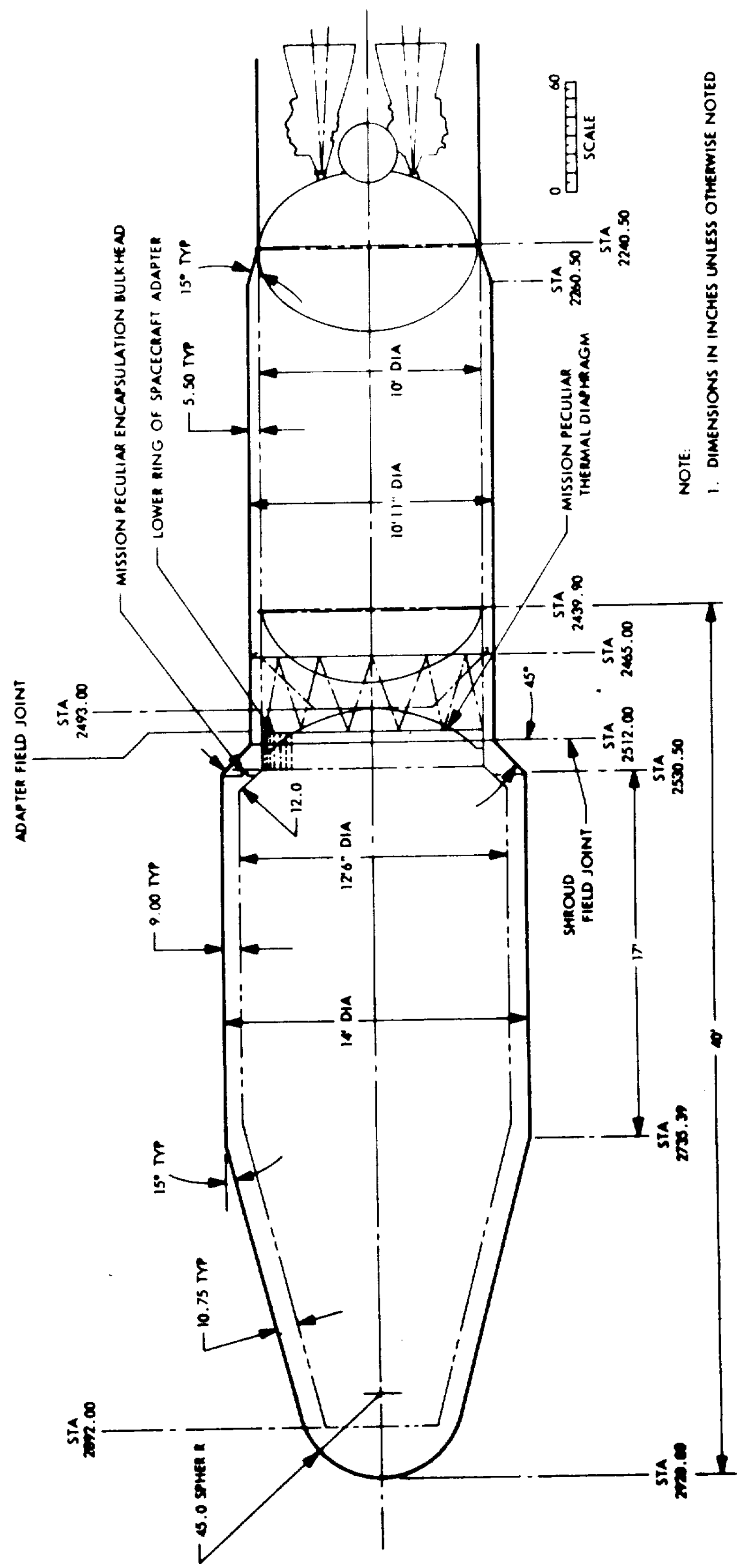

Figure 5F-2. Enshrouded Centaur

a 4-stage launch vehicle. The second contingency may be relaxed, of course, when pad loading schedules are negotiated and firm committments obtained. The third contingency is based on the fact that previous lunar and planetary mission experience has been with the 2 1/2 stage Atlas/Centaur and Atlas/Agena and not with a 4-stage vehicle.

The Integrate/Transfer/Launch (ITL) facility permits the liquid stages of both launch vehicles to be accommodated simultaneously in the vertical integration building (VIB). When the first vehicle is ready, it is transported by rail to the solid motor assembly building (SMAB), where the two solid motors are attached, and then to launch complex 41. The spacecraft can be mated to Centaur either in the VIB or on the pad insofar as the ITL facility is concerned. Mating on pad, however, allows a maximum de-coupling of spacecraft and launch vehicle checkout operations until each is ready for combined systems tests. Some simplification in mission peculiar OSE/AGE is also expected in this mode because on-pad checkout is required regardless of the checkout that would have been done previously in the VIB. Also, the prior encapsulation of the spacecraft in the nose fairing eliminates imposition of spacecraft cleanliness requirements on the universal environmental shelter (UES), which is located at the top of the mobile service tower.

Mission peculiar AGE requirements include airconditioning, S-band and VHF antennas, DC and 400-Hz power, an umbilical cable, video pairs for science checkout, antenna switching, and numerous hardline monitor and control provisions for the spacecraft. The magnitude of the airconditioning requirement will depend on whether RTG's are used in the lander. Maximum electrical power dissipated in the lander and orbiter is about 680 watts. AGE requirements also include instrumentation provisions as required to measure the acoustic environment external to the nose fairing during lift-off.

3. Nose Fairing

The nose fairing is assumed to be of aluminum, stiffened skin construction, and designed to permit total encapsulation of the spacecraft prior to mating with the launch vehicle. The diameter available as spacecraft dynamic envelope is assumed to be 12.5 ft. A somewhat larger diameter is desirable, however, to accommodate any changes in spacecraft configuration at the time a detailed design is begun. Because the launch vehicle adapter is approximately 10 ft in diameter, an annular bulkhead in the lower area of the nose fairing is required to close the gap between the isolation diaphragm on the adapter and the nose fairing.

Other than for airconditioning, no access doors for the spacecraft are required in the nose fairing. Positive closure of the airconditioning door can be assured through such a mechanism as was provided in the MV67 umbilical door in the Agena nose fairing. If RTGs are used in the lander, air ducts will be required in the nose fairing to direct cool air to selected areas on the lander. Duct locations are assumed to be in areas where ample clearance with the spacecraft exists.

The nose fairing/insulation panel assembly can be designed for jettison in either the Titan phase of flight or Centaur phase, but not both. As noted earlier, the resulting trade-off affecting the spacecraft is between launch vehicle performance and spacecraft cleanliness. Current LERC plans are to eject the assembly during Titan phase.

4. Interface with the Spacecraft

The structural interface with the launch vehicle is illustrated in Figure 5F-3. The encapsulated spacecraft assembly has a bolt circle field joint with the launch vehicle at the interface between the lower ring of the spacecraft truss adapter and the upper ring of the launch vehicle truss adapter. GD/C drawings PD68-0173 and SKC-9 were used as guides. The checkout and encapsulation of the spacecraft will be done in a spacecraft checkout facility, and the encapsulated assembly mated to the launch vehicle on pad or in the VIB as determined later.

Integral with the spacecraft adapter is an isolation, or "thermal", diaphragm attached to the lower ring. Closure of the nose fairing cavity is completed by an annular bulkhead between the diaphragm and the nose fairing, as discussed earlier. Whether this bulkhead incorporates airconditioning ducts is not currently established.

A temporary structural attachment is required between the nose fairing and the lower ring of the spacecraft adapter to provide support for the

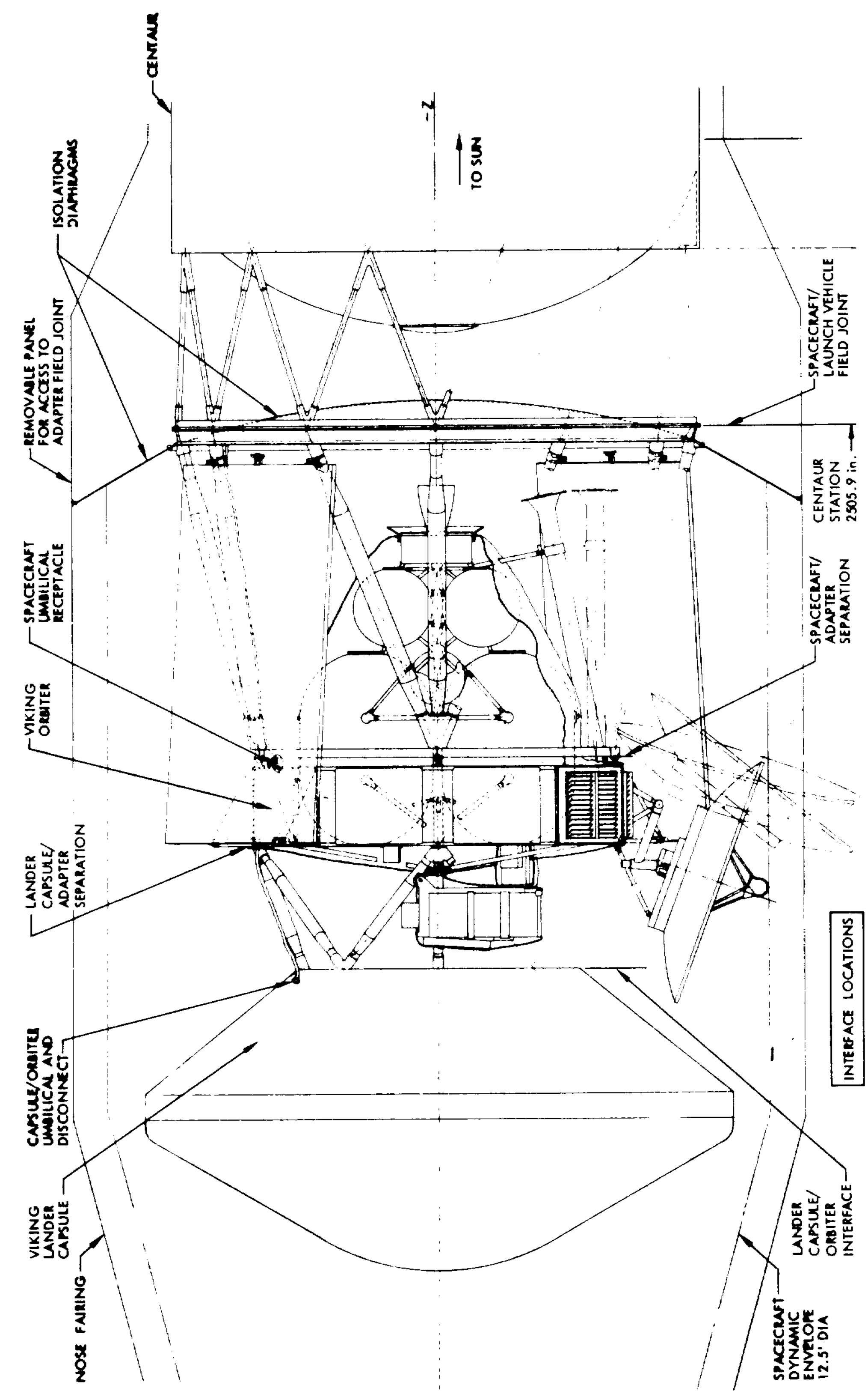

Figure 5F-3. Spacecraft/Launch Vehicle Interfaces spacecraft during hoisting operations. Another reason for this structural attachment is that, in hoisting the encapsulated assembly to the top of the Centaur stage, the handling sling must include a direct connection to the spacecraft adapter for safety reasons.

A single electrical and RF umbilical cable harness is assumed for the lander and the orbiter. The inflight disconnect is located at the upper ring of the spacecraft adapter. The connector to the external umbilical cable is a female fitting located in the Centaur adapter below station 2505.9. The number of coaxial cables in the umbilical are as required to checkout the lander and orbiter RF systems independently via low-loss coaxial cables to power monitors in the appropriate checkout areas of LC41, and by air link for checkout of the data streams.

Airconditioning of the nose fairing cavity is required for both the orbiter and the lander, as discussed earlier. In the launch configuration the solar panel tips are snubbed to the spacecraft adapter rather than to the lander. One reason for downward orientation of the panels is to provide free convection of conditioned air to the aft side of the lander.

Prior to nose fairing separation, orbiter composite telemetry modulation, including lander modulation when present, is transmitted via the launch vehicle telemetry system. This two-wire interface is incorporated in the inflight umbilical disconnect. This electrical interface also includes a circuit from the Centaur programmer which initiates separation of the spacecraft after injection velocity is reached, any orientation maneuvers are complete, and angular motion transients of Centaur have subsided. Spacecraft separation is verified via Centaur telemetry as sensed by microswitches or other devices.

Mounted on the spacecraft adapter are transducers of wide dynamic range for measuring stresses indicative of spacecraft axial, shear, and torsional loads from lift-off to separation. Transducers are also provided for measuring nose fairing cavity pressure and temperature. Appropriate amplifiers or signal conditioners are mounted nearby.

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5. Spacecraft Environment

Because the Viking Titan Centaur has not flown, the dynamic load environment for spacecraft design and testing must be inferred from Titan flight records and from dynamic aeroelastic analyses using stochastic exitations derived from Titan and Centaur flight histories. Vibratory loads, however, can be approximated from Centaur environmental design and test requirements given in GD/C report 55-00200E.

Acoustic input to the cavity of the Titan IIIC nose fairing is shown in Figure 5F-4. For Viking, the external exitation at lift-off should change very little from Titan IIIC; whereas at Mach 1 and max q the external environment may change considerably due to differences in pressure fluctuations (buffeting) as a function of bulbous nose fairing aerodynamics. Changes in internal acoustic environment will also change due to differences in nose fairing attenuation and in structural attachment. Shock loads will depend primarily on fairing and spacecraft separation system design.

6. Launch Vehicle Performance

The purpose here is to provide an estimate of launch vehicle mission peculiar performance. The end result of interest is the available launch period as predicated on optimal launch dates, other parameters being both fixed and conservative. The assumptions made for the Viking mission are listed below, and the corresponding performance is shown in Figure 5F-5.

Parking orbit altitude (n. m.)	90
Maximum Centaur coast time (minutes)	60
Minimum launch azimuth (deg)	45
Maximum launch azimuth (deg)	115
Spacecraft support weight (lbs)	395
Centaur jettison weight (lbs)	4178
Centaur performance margin (% ΔV)	2.5 (3σ)
Expansion ratio of Titan Stage I engines	15:1
Nose fairing height above field joint (ft)	34
Nose fairing skin diameter (ft)	14

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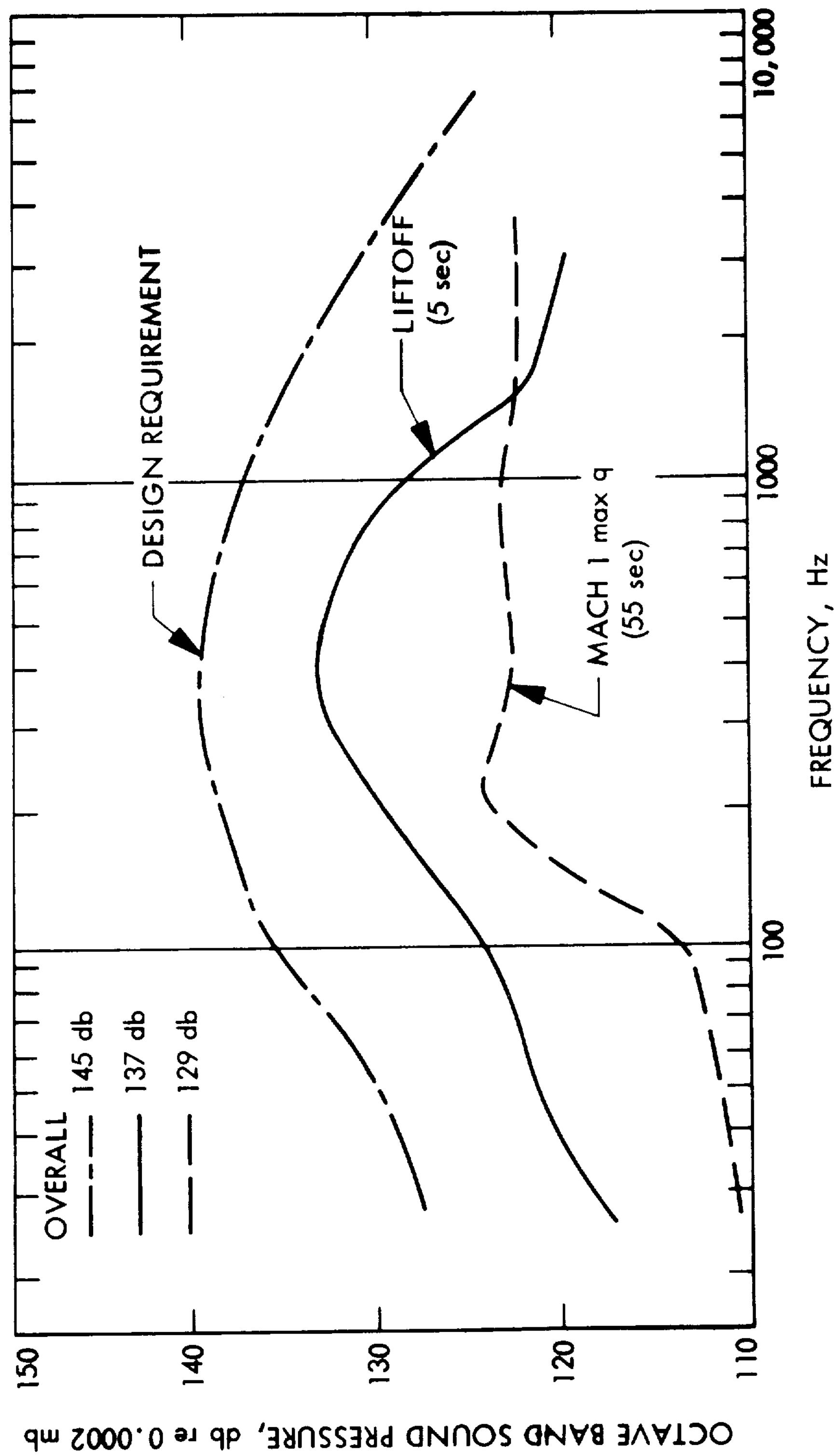

Figure 5F-4. Spacecraft Acoustic Loads for Titan IIIC

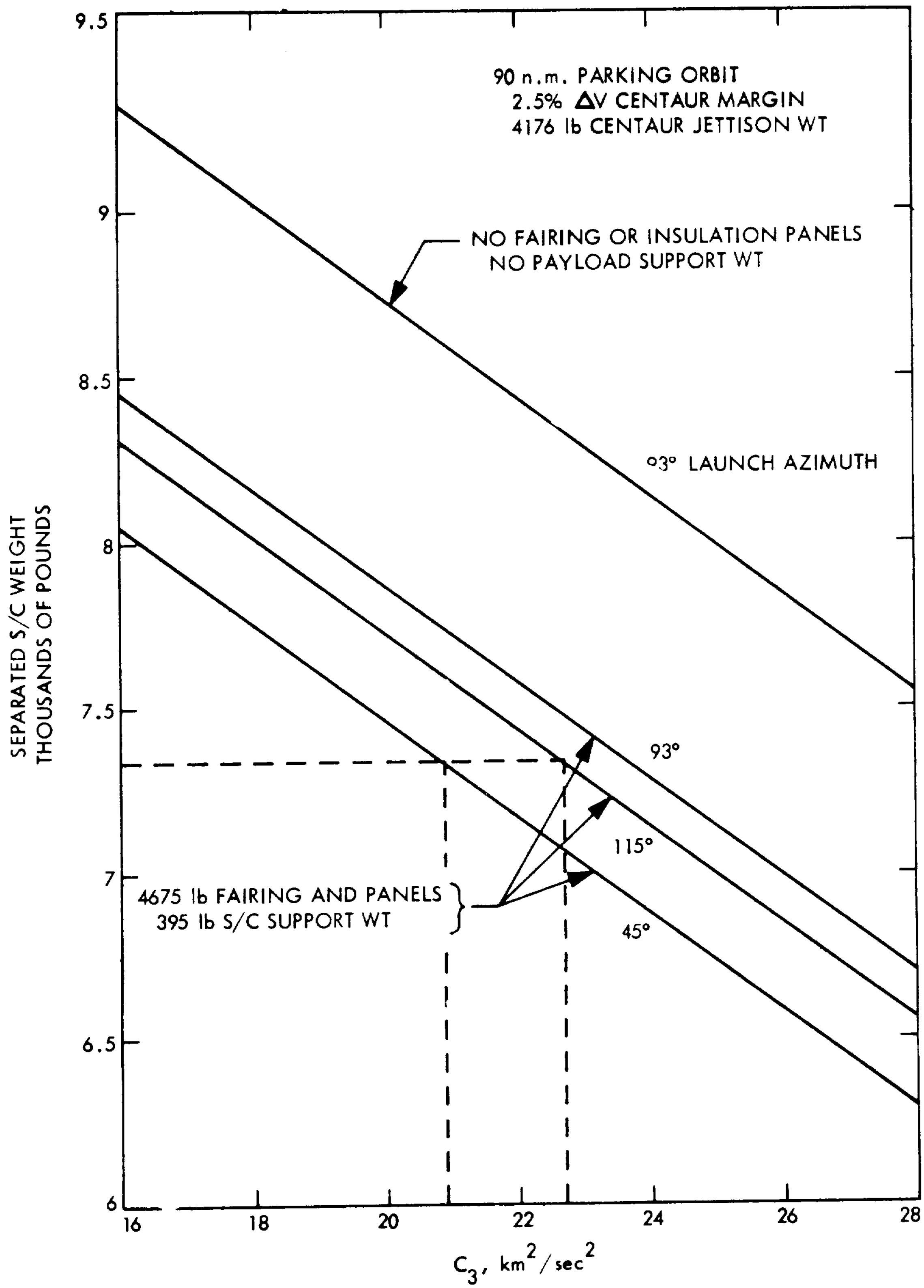

Figure 5F-5. Launch Vehicle Performance for Viking Mission

Nose fairing and insulation panel weight (lbs)		4675
Drag loss due to bulbous fairing (lbs)		100
Fairing exchange ratios:	$C_3 = 17.1$	0.08
	$C_3 = 27.3$	0.075

No performance reserve is shown for Titan because it burns to depletion. The nose fairing skin diameter of 14 feet is assumed to have a dynamic clearance envelope for the spacecraft of 12.5 feet or slightly more.

Based on Figure 5F-5, a 7335 lb spacecraft can be injected into a transfer orbit at a maximum C_3 of 20.9 km^2/sec^2 for a launch azimuth of 45 degrees, and 22.7 km^2/sec^2 for a launch azimuth of 115 degrees. When these constraints are coupled with the planet arrival constraints of the lander and the maximum approach velocity constraint of 3.3 km/sec, the resulting launch periods are 40 days minimum for a 45 to 115 degree launch azimuth corridor, and 20 to 30 days minimum for a 65 to 115 degree launch azimuth corridor.

7. Interface Control

Because mission success requires careful control of functional interfaces between major subsystems, the number of organizations involved in each interface is correspondingly important. Although methods of interface control are outside the scope of this document, it is useful to examine the surprisingly large number of organizations and associate contractors involved. A partial listing follows:

Langley Research Center	Viking Project
Lewis Research Center	Launch Vehicle
Jet Propulsion Laboratory	Viking Orbiter
USAF/SAMSO	Launch Vehicle
Aerospace Corporation	Launch Vehicle
6555 Air Test Wing	Launch Operations
Kennedy Space Center/ULO	Launch Operations
Martin-Marietta Corp.	Launch Vehicle
General Dynamic-Convair	Launch Vehicle

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Pratt & Whitney	Centaur engines
Minneapolis Honeywell	Centaur guidance
UTC	Titan engines

In addition to these will be the Viking contractors yet to be selected, including their associate contractors.

Interface working groups will provide the direct contacts needed to insure compatible interfaces; but because the various agencies and contractors do not have identical internal organizational structures, considerable effort is required in developing charters for the working groups which do not conflict with each other. The fact that schedules are basically dictated by the Mars ephemeris allows little margin for correcting incompatibilities between systems. Interface test schedules, as well as combined systems tests at ETR, will reflect this rigorous time constraint.

SECTION VI

BASELINE ORBITER MISSION DESCRIPTION

A. ORBITER MISSION PROFILE

The following subsections describe an orbiter mission profile consistent with the conceptual design and interpreted from the guidelines and rationale from LaRC. Each subsection summarizes an orbiter peculiar mission phase. The assumptions and detailed mission sequence of events are given in section VII. C.

1. Launch, Midcourse, and Interplanetary Cruise

These phases are important but not peculiar to the Viking mission and, as such, are not summarized. They can be considered similar to corresponding phases of any Mariner derivative spacecraft such as MM69 or MM71. The details of these phases are given in the sequence in section VII. C.

2. Orbit Insertion

The orbit insertion maneuver consists of a series of spacecraft turns to orient the thrust axis for the burn, a verification of this orientation by command from Earth, the burn, and the turn unwinds to reacquire the celestial references.

Figure 6A-1 illustrates the multiple maneuvers required to attain the burn attitude with the VLC aboard. The standard roll and pitch maneuvers orient the V S/C thrust axis for the burn. The requirement for attitude verification requires engineering telemetry during the maneuver period. However, with the VLC aboard the forward hemisphere of the V S/C is not covered completely by a low gain antenna. The maneuver antenna on the end of a solar panel must be reoriented to the quadrant of the hemisphere containing the Earth. Engineering telemetry and command capability results. This is accomplished by a second roll turn to view Earth from the antenna. The thrust axis orientation is not changed by this roll turn because the roll axis is the thrust axis

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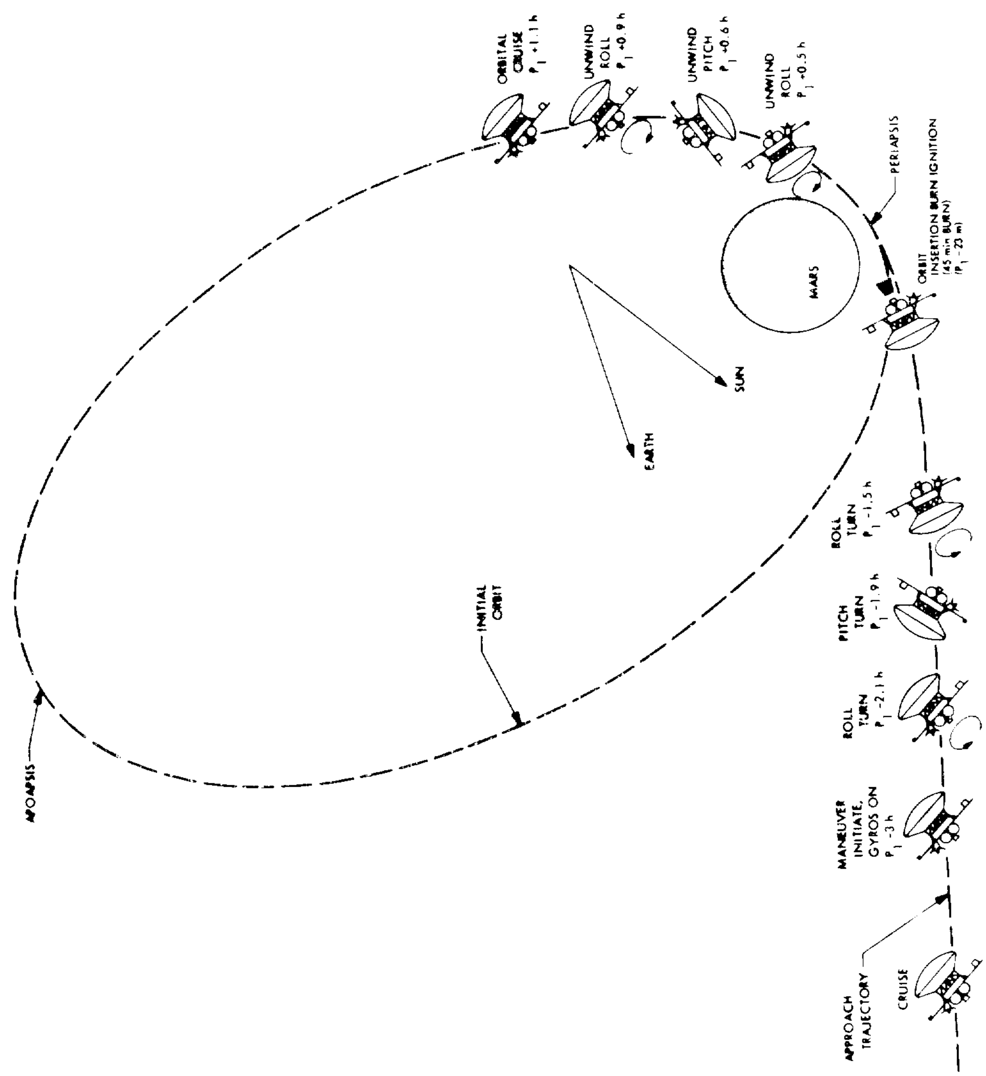

Figure 6A-1. Viking Flight Sequence -- Orbit Insertion in this VO design. Receipt of the engineering data and other information about the orientation of the spacecraft results in an MOS decision to inhibit or begin the insertion burn. The burn duration with the single engine design concept lasts approximately 45 minutes. Upon termination of the burn, the VO begins the unwind maneuvers to reorient the spacecraft on the celestial sensors for the first orbit cruise and battery charging.

This phase of the mission sizes the (VO) battery. The extended period off the Sun line for maneuvers and attitude verification as well as the power required for the 45-minute burn is the largest watt-hour requirement for the mission. This will be discussed in section VII. E.

3. Preseparation Surveillance Phase

Following orbit trims to provide the desirable period and periapsis altitude, planet surveillance begins to locate a landing area for the VL. The surveillance utilizes VO science to acquire topographical, thermal, and physical data at proposed landing longitudes. Landing latitudes are somewhat restricted at this time to within the de-orbit capability of the VLC propulsion subsystem.

Since all of the VO science is boresighted by the science platform, it is necessary to orient the platform to allow a near periapsis view period of approximately one half hour. However, with the VLC aboard the science platform is restricted in its cone angle travel. The scan limits required for near periapsis viewing of Mars were identified early in the study based on a MM71 location of the science platform (i.e. on the VO roll axis).

For this conceptual design, the decision was made to retain the MM71 science platform location. To circumvent the view restrictions caused by the VLC aboard, a spacecraft maneuver was derived to allow unrestricted viewing of the planet surface at periapsis passage. This entails a two-turn maneuver to orient the platform clock articulation plane parallel to the plane of the orbit. Figure 6A-2 illustrates these turns. An inertial hold maintains the final V S/C orientation while the science platform slews in clock only to map the planet. Communications are maintained through the low gain antenna pointing in -z direction. All science is recorded. (A second roll turn may be necessary to orient the science platform within its clock angle structural constraints.) At science termination, the VO begins an unwind maneuver to reacquire

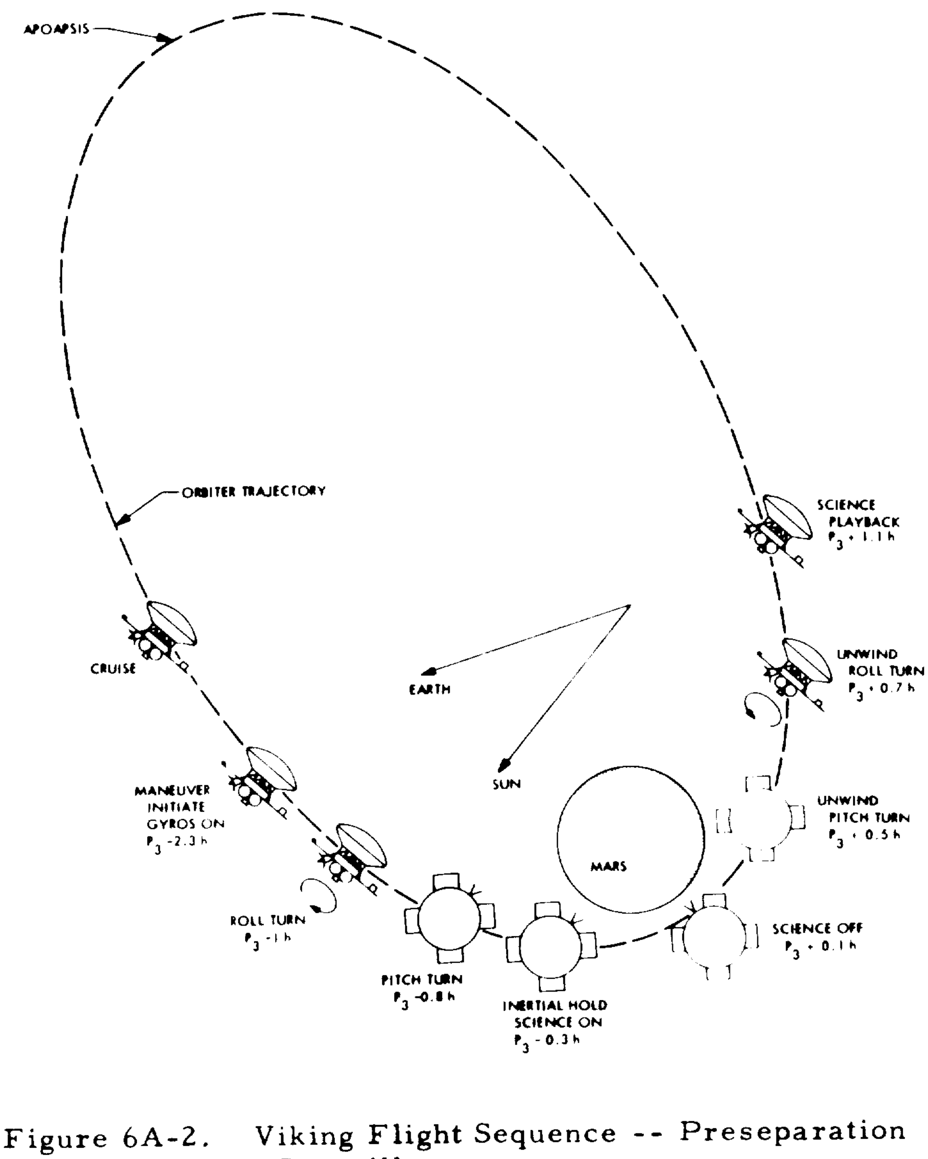

Figure 6A-2. Viking Flight Sequence -- Preseparation Surveillance celestial references. With Sun and Canopus acquired, the VO telemetry subsystem begins playback of the recorded science and battery charging begins.

This is typical of a mission phase to be repeated periodically until the landing site is chosen from the processed science.

4. Capsule Separation and Tracking Phase

Orbit trims are performed to place the spacecraft in a Mars synchronous orbit over the desired landing site. A period of two days is then required to interrogate the VLC before separation.

Figure 6A-3 shows the key events in the VLC separation orbit. The VLC separation sequence begins by performing a pitch turn to orient the spacecraft for the bioshield cap jettison event. A single pitch turn is sufficient to place the jettison direction away from the planet to a higher energy orbit. This prevents possible communications interference later in the mission and insures planetary quarantine. Unwinding the pitch turn, to reacquire celestial references, is followed immediately by the VLC separation maneuver.

A mission constraint imposed on the orbiter is to orient the spacecraft to the VLC burn attitude before VLC separation. A maneuver consisting of a pitch and a roll turn provides this attitude. High rate engineering telemetry is acquired and recorded from the VLC during and immediately following the separation event. Due to configuration restrictions, the RF relay link antenna (antenna L) on the VS/C utilized immediately following capsule separation is contained within the bioshield base and is provided by the VLC. Communications are maintained until termination of the VLC burn.

VLC communications are interrupted while the VO unwinds to reacquire celestial references. Following reacquisition, the VLC adapter is jettisoned while on the Sun line to provide unobstructed relay communications through the VO antenna.

The VO reacquires VLC communications before entry for interrogation and pre-entry engineering data. The RF relay data link is established between the VLC, VO and DSN.

VLC entry data is fed through the VO in real time during the entire entry phase.

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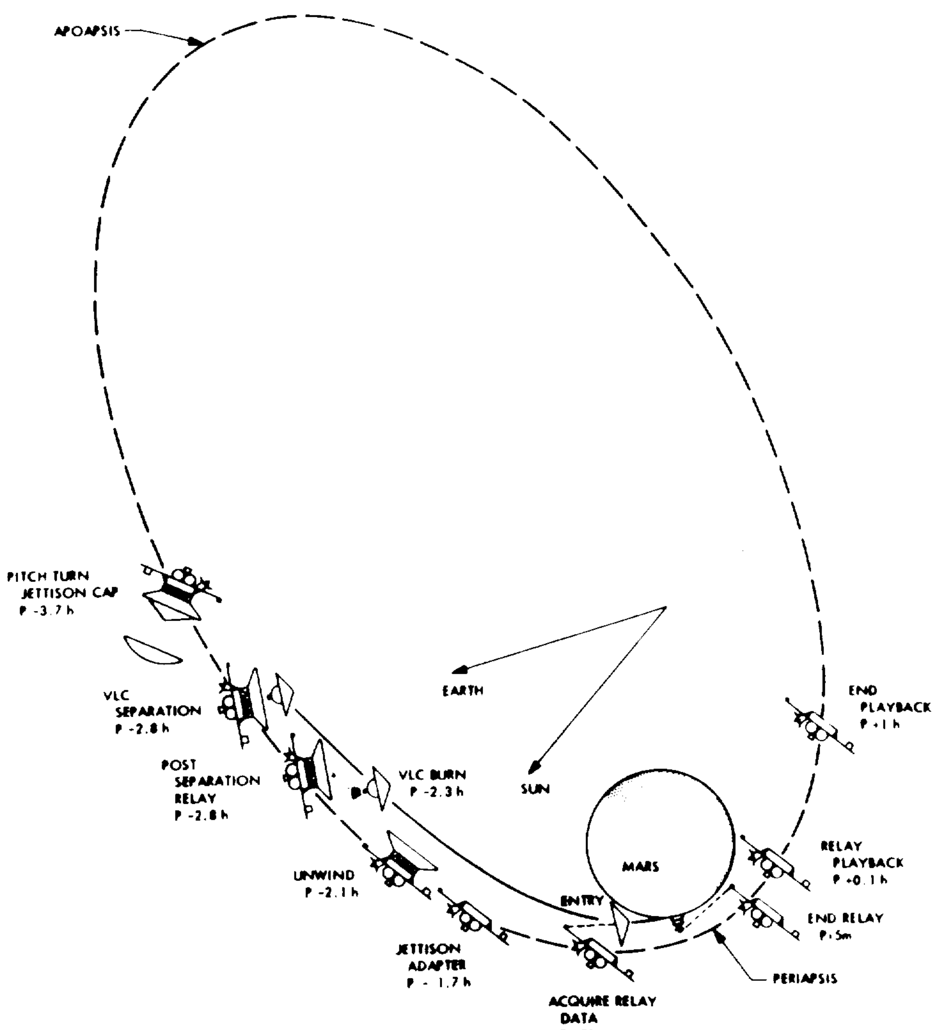

Figure 6A-3. Viking Flight Sequence -- VLC Separation

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At VL touchdown the telemetry modes in the VO and VL are changed to acquire landed science. Initial high rate VL data is received and recorded in the VO.

Following periapsis passage and lander communications, the VO telemetry subsystem initiates playback of the recorder. Pre-entry VLC engineering telemetry that was recorded and the initial VL landed science are relayed to the DSN.

Subsequent passes acquire landed science and are performed with the VO oriented on the celestial references (no maneuvers required).

5. VL Relay Passage

A series of VO-VL relay passes follows the initial landing pass to acquire high-rate science data. The following paragraph describes a typical pass.

As the VO approaches the VL, the beacon transmitter on the VO broadcasts a signal to start the VL relay radio. Once the RF link is established, the VL begins high-rate science readout to the VO. The VL data is recorded by the VO throughout the pass. Playback begins after termination of the VL signal. This phase is illustrated in Figure 6A-4.

This phase will be repeated for three periapsis passes following separation in the nominal mission. However, if the VL direct communication link fails this phase may continue. Following the nominal three passes, an orbit trim will occur to resume broad planet surveillance by the VO science. A mission constraint requires the VO to pass over the VL once per month for communications until the end of the mission.

6. Orbiter Science Passage

Subsequent to the relay communication and orbit trim phases, the VO will begin a phase of broad planet surveillance. The non-synchronous surveillance orbit will allow a progression of science swaths at each periapsis passage. A typical pass is described below.

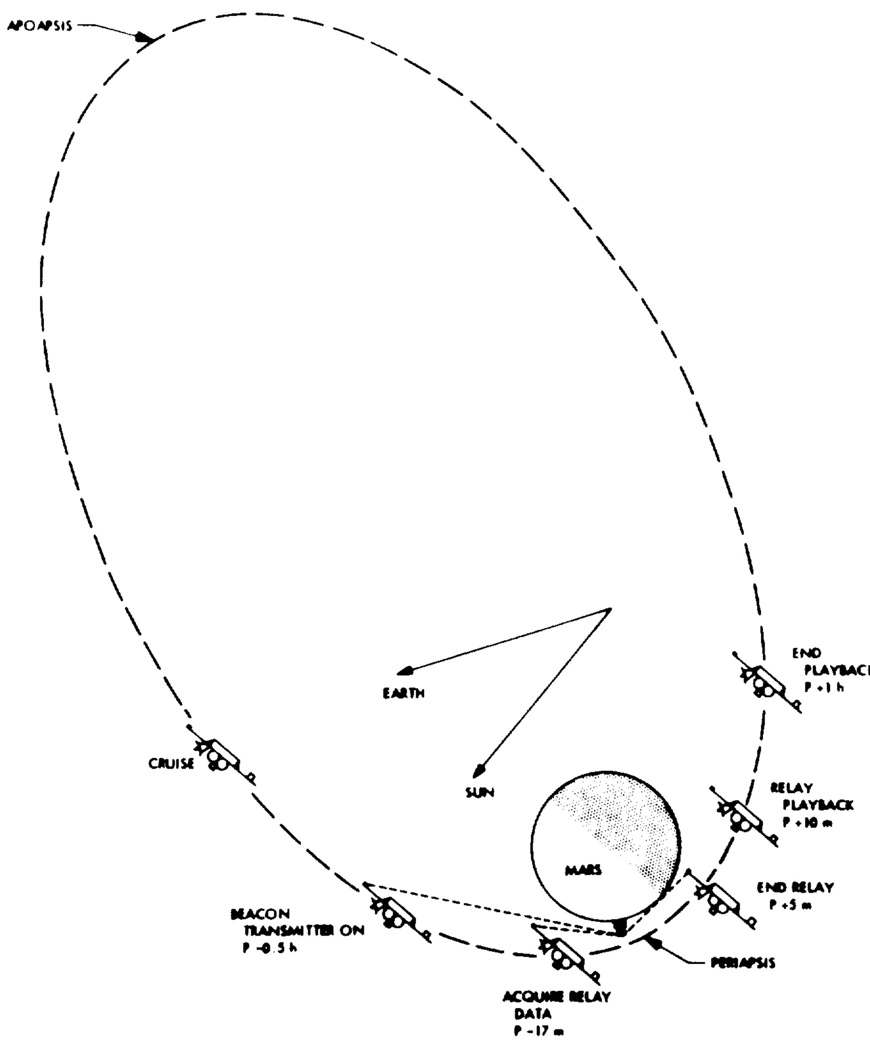

Figure 6A-4. Viking Flight Sequence -- VL-VO Relay Passage

The VO configuration for these science passes is similar to the MM71 configuration. The science instruments boresighted by the articulating science platform can be pointed, within structural constraints, anywhere within a large portion of the +z direction hemisphere. (The absence of the VLC allows large cone angle variations.)

Figure 6A-5 illustrates a typical pass. It begins by slewing the science platform in cone and clock to the first science acquisition direction. Preprogrammed sequences aboard the VO allow articulation of the science platform to update pointing throughout the passage. When the science platform is oriented correctly, science begins and is recorded. Repetitive TV frames or other science continue until the capacity of the recorder is reached or the science is terminated. Playback of the recorded VO science begins as soon as the DSN is in view.

This phase is repeated for the remainder of the mission with the exception of VL relay passes when required.

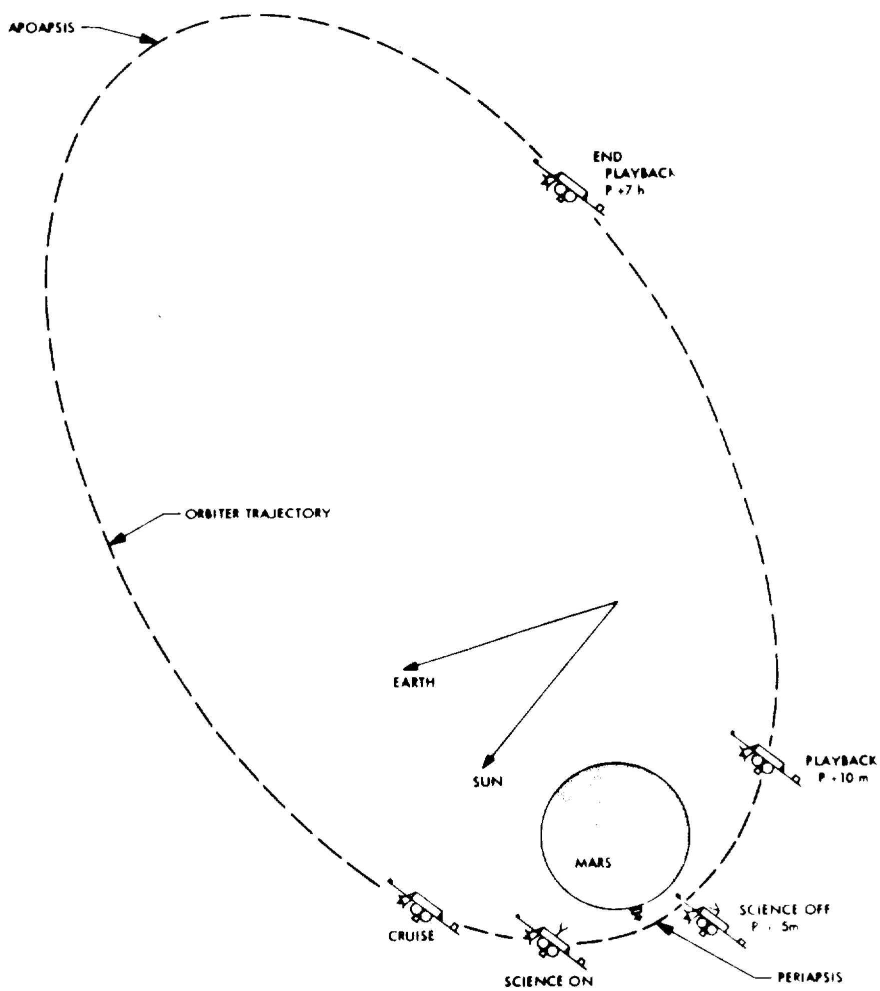

Figure 6A-5. Viking Flight Sequence -- VO Science Passage

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B. SCIENCE DATA RETURN ASSUMPTIONS AND CONSTRAINTS

1. Pre-separation Site Survey

A minimum of 1.6×10^8 bits of data from the orbiting spacecraft taken in each potential landing site area of 300 km along the orbit path by 100 km normal to the orbit path is required. Most of this data is TVS Camera B data.

A nominal requirement is 3.2×10^8 bits of Camera B video data from each potential landing site, or the amount of data in an array of 4 x 15 pictures covering the footprint at periapsis.

2. Orbiter Data After Capsule Landing and Support

After three days or more in which lander data is relayed to the DSN, orbital science operations will begin on a non-interference basis. TVS and other data will be recorded each orbit as permitted by lander relay operations. During each orbit, when landed data is not relayed, a set of TV pictures is taken near periapsis. Recorded pictures may be played immediately to the DSN except at the end of the Canberra dish operational period (see Figures 6B-1 through 6B-4), or replayed after Goldstone rise and prior to the next photo sequence.

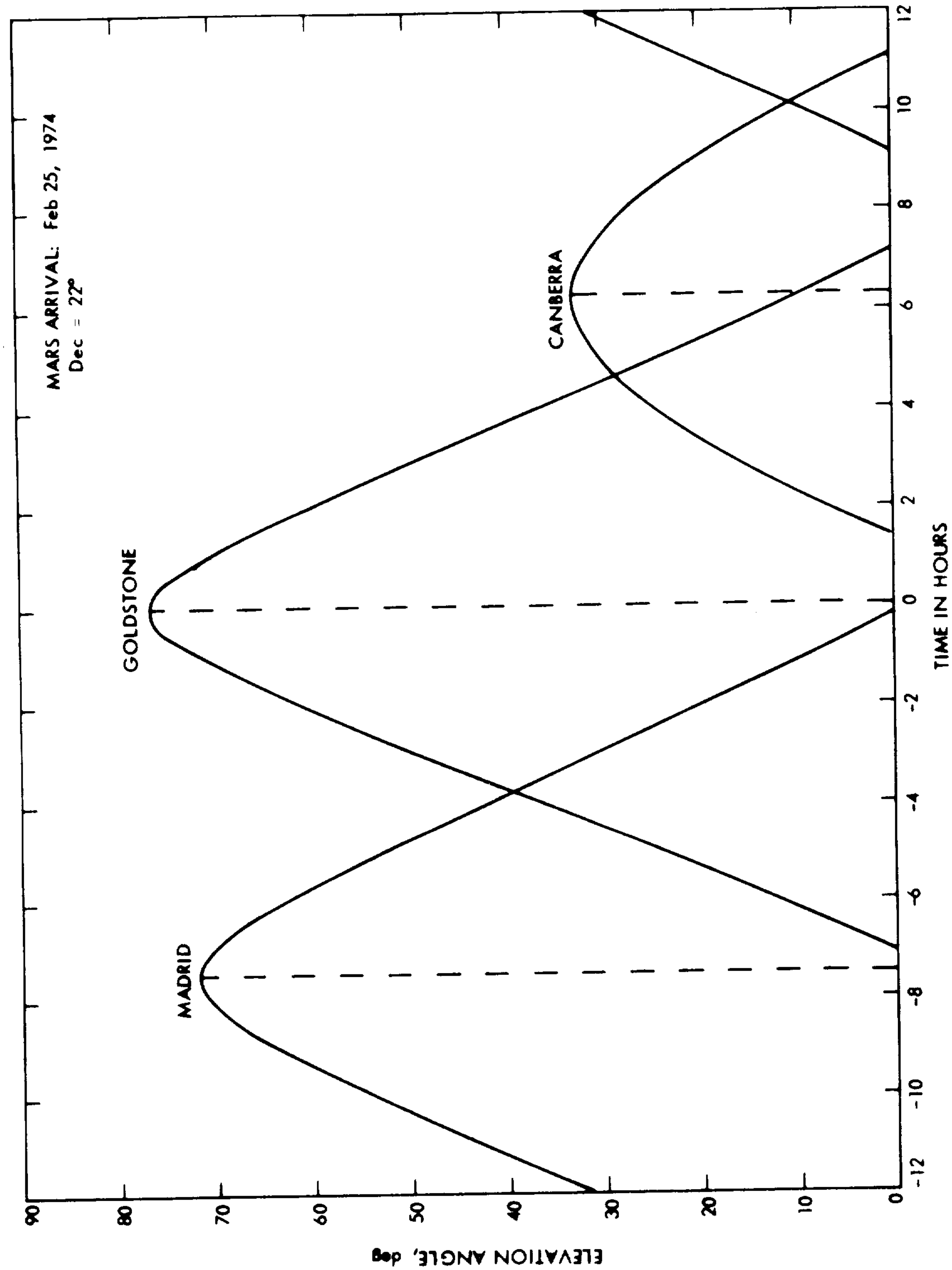

Figure 6B-1. 210' Antenna View Periods for February 25, 1974 Arrival

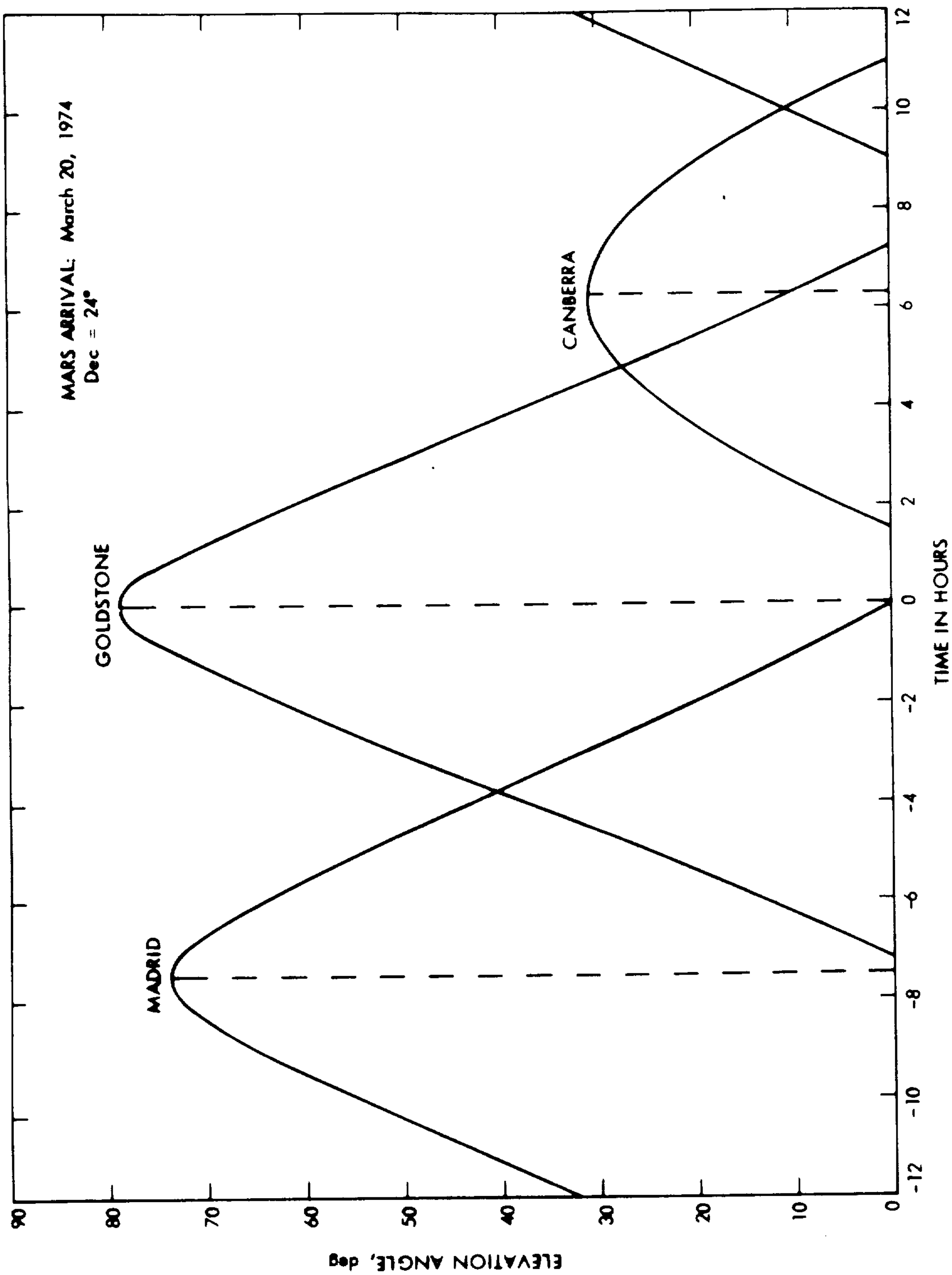

Figure 6B-2. DSN 210' Antenna View Periods for March 20, 1974 Arrival

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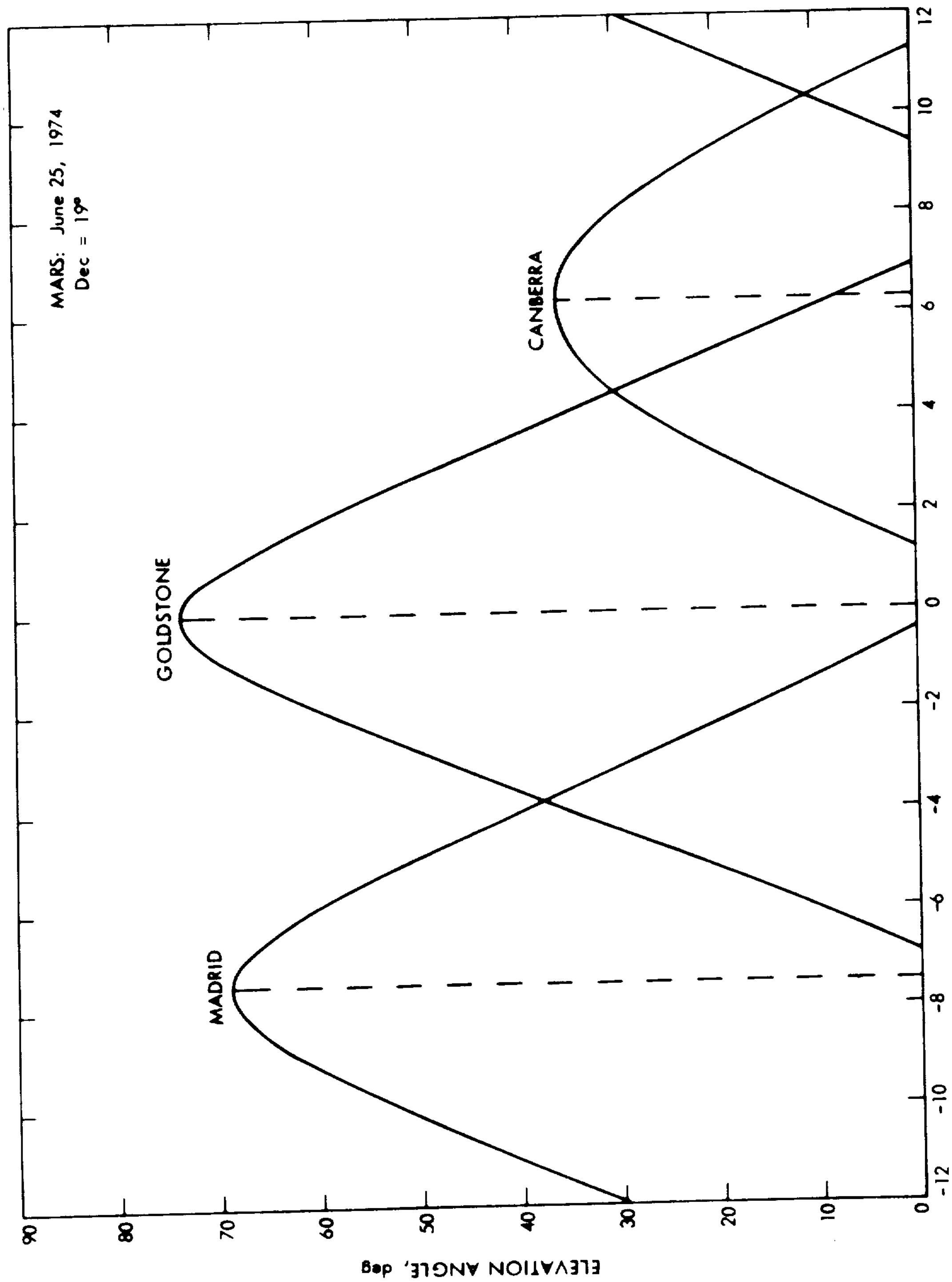

Figure 6B-3. DSN 210' Antenna View Periods for June 25, 1974

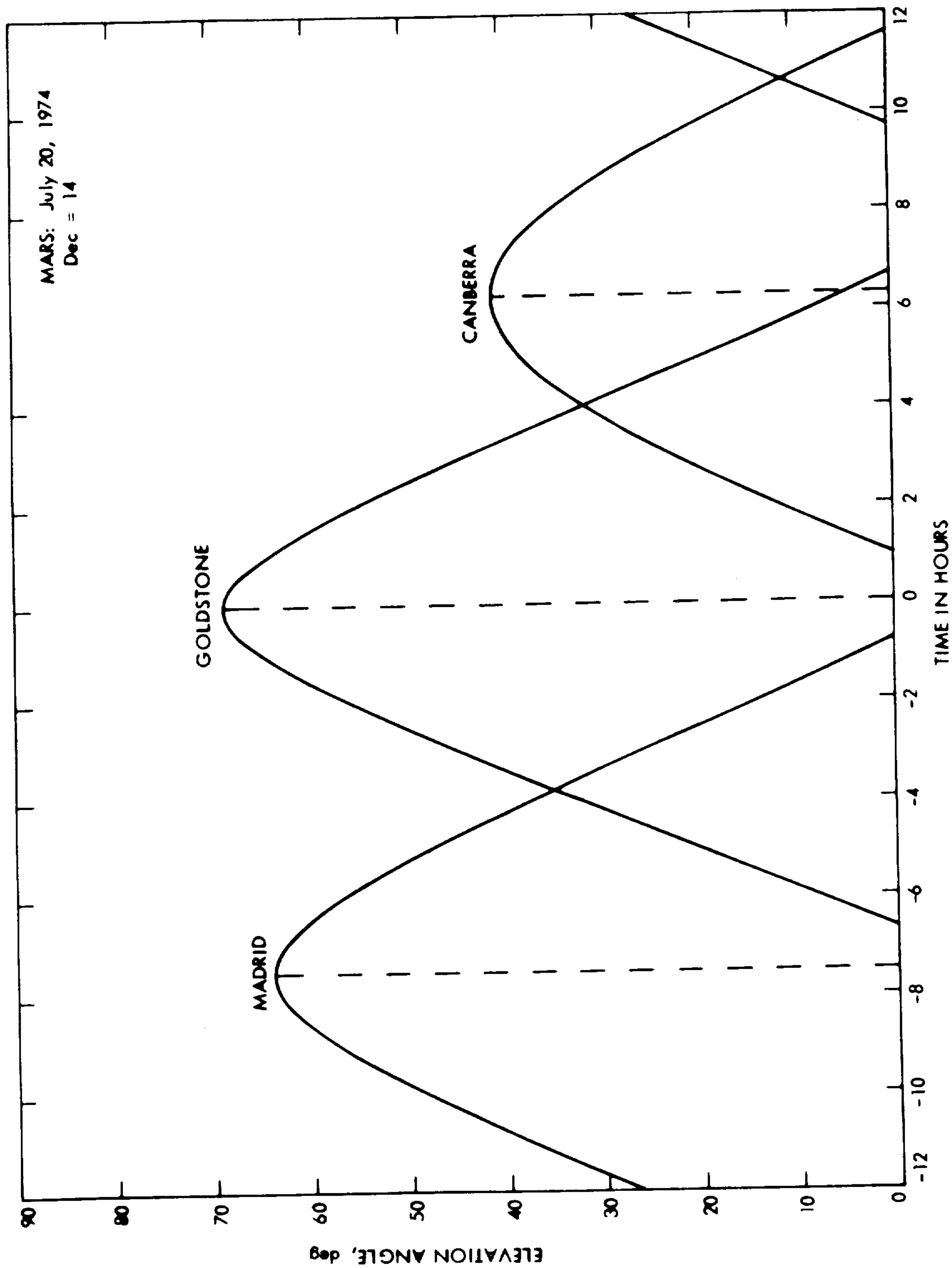

Figure 6B-4. DSN 210' Antenna View Periods for July 20, 1974

SECTION VII

BASELINE VIKING ORBITER DESCRIPTION

A. INTRODUCTION

It is the purpose of this section to outline in as much detail as possible, within the time and scope devoted to this conceptual design, the individual subsystems which as a composite represent the orbiter portion of the Viking spacecraft.

The Viking orbiter design is based on Mariner technology, and on the Mariner '71 spacecraft in particular. The actual details of the subsystems which are represented in the Viking spacecraft are described in detail in the following section.

B. ORBITER CONFIGURATION

1. Constraints and Guidelines (Mechanical)

1) The maximum orbiter weight is 5362 lbs excluding a lander capsule adapter.
2) Orbiter science is assumed to consist of television, IR radiometer, IR spectrometer, and a multi detector IR radiometer. Orbiter science payload weight will not exceed 150 lbs, including allowance for the SDS. All orbiter science instruments are mounted on the scan platform.
3) Launch vehicle is Titan/Centaur.
4) The "shrouded Centaur" concept with 12 point truss launch vehicle adapter as shown in GD/C drawing PD68-0173, is the baseline nose fairing concept. The dynamic envelope is 12.5 feet in diameter and assumed to be at least 18 feet long.
5) The nose fairing and launch vehicle adapter are designed to permit S/C encapsulation prior to mating with the launch vehicle, and include isolation diaphragms. On-pad air conditioning of the nose fairing cavity will be provided.

   The spacecraft umbilical connector will be located near the forward end of the Centaur.
6) The lander capsule will be mounted above the orbiter in the launch configuration, and will be on the shady side of the orbiter during cruise.
7) The lander capsule weight at launch is 1800 lbs maximum, excluding its adapter.

   The mechanical interface between the lander capsule and orbiter will be an assembly joint attaching to the lander capsule adapter at 8 points on a 62-inch diameter. The lander capsule internal structure will be designed to accommodate support loads applied by the 8 point truss adapter.

The lander capsule bioshield cap and base sections will be jettisoned in orbit.

Umbilical functions are required to support the lander. The lander capsule CG is nominally on the roll axis, 38.4 in. from the VLC/orbiter interface. The lander capsule moments of inertia are:

a) At launch

$I_{zz} = 418 \text{ slug ft}^2$ (JPL coordinate axes)

$I_{xx} = 275 \text{ slug ft}^2$ (JPL coordinate axes)

b) Lander capsule less bioshield cap

$I_{zz} = 370 \text{ slug ft}^2$ (JPL coordinate axes)

$I_{yy} = 252 \text{ slug ft}^2$ (JPL coordinate axes)

The lander is separated from the orbiter after Mars orbit is achieved.

8) Design of the orbiter will be constrained to using 1969 state-of-the-art technology; Mariner '69 and Mariner '71 subsystem components will be used where possible.

2. Viking Orbiter

a. Mechanical Configuration -- Baseline Description

The configuration concept illustrated in Figure 7B-1 is designed to satisfy the constraints and guidelines noted in the previous subsection. Wherever feasible and possible, Mariner '69 and '71 components and technology are utilized.

The configuration concept is depicted in its launch configuration within the Titan/Centaur bulbous shroud. As shown, the entire spacecraft is encapsulated within the shroud and at the launch vehicle-spacecraft adapter interface by the isolation diaphragm and annular bulkhead. It should be noted that the shroud, Titan/Centaur field joint, isolation diaphragm, and annular bulkhead are only shown pictorially.

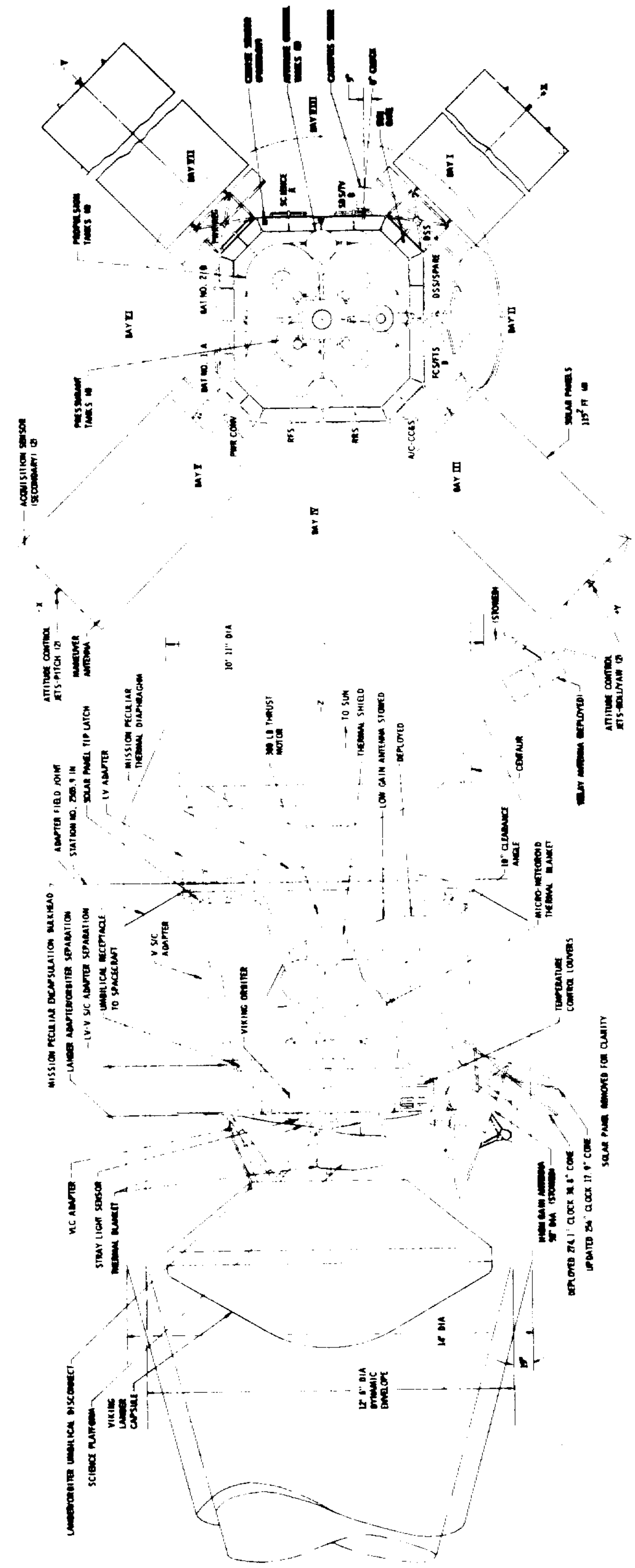

Figure 7B-1. Viking Spacecraft Configuration

The configuration of the Viking orbiter was significantly influenced by a bi-propellant propulsion system weighing approximately 4100 lbs -- more than three times the propellant volume of Mariner '71. The propulsion system utilizes the Mariner '71 300-lb thrust motor and related hardware where applicable. The four propellant tanks are cylinders with hemispherical ends and are 44-in. long x 30-in. diameter. It is planned to use the hemispherical forgings of the Mariner '71 for the ends of the propellant tanks. The four 18-in. diameter pressurant tanks are attached to the four propellant tanks with gimballed thrust motor mounted on them. The structural concept shown for the propulsion system enables it to be assembled as a separate entity and assembled to the octagon bus in modular fashion.

The orbiter bus was conceived as a "wrap-around" octagon structure. That is, the sizing of the bus was dictated by the size of the propulsion system and it was determined that an unequal-sided octagon bus wrapped around the propellant tanks had several significant advantages attendant to this concept. The size of the bus, 98" x 84" across the diagonals, is the smallest (therefore lightest) size required to carry the necessary orbiter electronics and other subsystems. A bus 98" x 84" allows up to 12 bays of standard Mariner electronic subsystem modules and chassis. This packaging scheme gives 4 more bays than the standard Mariner octagon bus and the added packaging volume is deemed to be sufficient to meet subsystem requirements. Nesting the propellant tanks within the bus adds to the protection of the propellant tanks from meteoroid penetration, enhances thermal control of the orbiter, and facilitates designing the structural support of the propulsion system to the bus in an efficient and modular manner. Louvers are attached to the bays on the sides of the bus to aid in thermal control of subsystem electronics.

Science instruments are mounted on a scan platform similar to the design used on Mariner '69 and '71. The platform will be mounted on the bus centerline in a manner which permits it to be deployed to the required cone and clock positions for scanning the planet. However, during the first days of orbiting the planet, while the lander capsule is still aboard, it will be necessary (since the platform cannot be deployed in cone) to maneuver the spacecraft to the proper inertial attitude and scan in clock only, in order to perform surveillance of possible landing sites. It is to be understood, depending upon the angular position of the platform, the truss members of the lander capsule adapter may be in the instruments view. However, it is felt that this will not seriously degrade the surveillance operation.

Solar panels are mounted to the bus in a fan-like array on the coordinate axes "in-line" with the propellant tanks to minimize any CG migration or offset problems on the attitude control system. The solar panels and cells are similar to the Mariner '71 except that the panel is a 48" x 86" rectangle. Four panels make up a total gross area of 115 sq. ft. which is sufficient to meet the power requirements of the spacecraft. As shown in the configuration, the panels are mounted to the bus on outriggers which are necessary to permit the high gain antenna to be deployed and pointed between solar panels during flight. At and during launch the panels are stowed in "sockets" to prevent any dynamic excursions of the panels. These sockets do not latch the panels and therefore the panels will "fly out" of them when the spacecraft is separated from the spacecraft adapter.

Attitude control jets for spacecraft pitch, roll, and yaw, coincident with the coordinate axes, are mounted at the outboard edge of each of the solar panels giving a moment lever arm of 155 inches from the roll or "Z" axis. Two 12-in. attitude control bottles are nested and attached to the four propellant tanks. Celestial sensors, comprised of Canopus sensor, cruise and acquisition sun sensors, sun gate, and a stray light sensor, are mounted to the appropriate sides of the bus and solar panels to meet the required fields of view of the instruments. Only the acquisition sun sensors are mounted on the solar panels. Canopus and stray light sensors are on the science platform side of the bus. Whereas, the cruise sun sensor, and sun gate are on the sun or thrust motor side.

To satisfy communication requirements, two low-gain antennas are used to give approximate spherical coverage during maneuvers, an array of 2 relay antennas is used to communicate with the lander, and a 58-in. high-gain antenna is mechanized to give the necessary Earth-directed position at planet encounter and during orbit. Three antennas require post-separation deployment. These are the high-gain antenna mounted to the bus at Bay II, the omni low-gain antenna mounted to the bus at Bay II, and the relay antenna array mounted to the +y solar panel. The other low-gain maneuver antenna is mounted to the +x solar panel. The high-gain antenna deployment occurs in subsequence to the solar panel deployment. It is then stepped to other positions about a fixed axis during the orbital operations phase of the flight.

A meteoroid shield is incorporated into the thermal control blanket which is attached to the propulsion module. The bus and science platform will be enclosed in a gold-coated Kapton blanket for thermal control.

The Viking spacecraft is supported during launch on a four to twelve-point truss adapter as shown in the configuration drawing. Separation of the spacecraft from the adapter will utilize the technique of explosive bolts and a spring separating mechanism. The lander capsule is attached to the orbiter bus by an eight to four-point truss adapter. This adapter and the aft bioshield base section will be jettisoned sometime after lander separation. This jettison event will also use explosive bolt/spring separating mechanism techniques.

b. Inertial Properties

The spacecraft and orbiter inertial properties are shown in Table 7B-1.

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Table 7B-1. Viking Inertial Properties

Reference Plane: Capsule/Adapter Interface

<u>Total Spacecraft (wet) - 7335 lb</u>

CG_X	=	.027" from roll axis
CG_Y	=	.167" from roll axis
CG_Z	=	32.148" from interface towards propulsion end
I_X	=	3864 slug ft^2
I_Y	=	3758 slug ft^2
I_Z	=	1654 slug ft^2

<u>Total Spacecraft (dry) - 3978lb</u>

CG_X	=	.040" from roll axis
CG_Y	=	.306" from roll axis
CG_Z	=	9.838" from interface towards propulsion end
I_X	=	2536 slug ft^2
I_Y	=	2502 slug ft^2
I_Z	=	1238 slug ft^2

<u>Orbiter (dry) - 2108lb</u>

CG_X	=	.076" from roll axis
CG_Y	=	.579" from roll axis
CG_Z	=	50.757" from interface towards propulsion end
I_X	=	585.7 slug ft^2
I_Y	=	551.3 slug ft^2
I_Z	=	803.0 slug ft^2

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C. ORBITER SEQUENCE

To provide the Jet Propulsion Laboratory technical divisions with a nominal mission outline, the sequence of events that follows was generated. The assumptions utilized are given below and include nomenclature, timing, orbit, subsystems assumptions, and periapsis passage guidelines. Table 7C-1 contains a table of sequence particular acronyms. Telemetry modes defined at this time are in Table 7C-2. An index of key mission events is also presented in Table 7C-3.

1. Nomenclature

The Viking terminology used in this sequence is tabulated in the Glossary and reflects the definitions presented to JPL by Langley Research Center.

A list of acronyms used in this sequence is presented in Table 7C-1.

2. Sequence Timing

1) All times are nominal and subject to change.
2) Viking Lander Capsule and Lander event times reflect guidelines from Langley Research Center.
3) Initial orbit phase MOS events and timing are taken from JPL mission operations guidelines.

3. Conceptual Orbit

1) Approximate period 24.6 hrs (Mars synchronous).
2) Periapsis altitude, 1000 km.
3) Other parameters reflect Langley Research Center guidelines.
4) Orbit insertion burn duration 2700 seconds.

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5) Orbit trim strategy -- the period between 1st periapsis (encounter) and first apoapsis shall be sufficient to provide orbit determination data. This allows the first surveillance orbit trim to occur at the second periapsis. Other trims may be required for preseparation surveillance purposes and to provide a Mars synchronous orbit prior to VLC separation. Following VLC separation and the subsequent communications periods, orbit trims will provide further planet surveillance and VL reacquisition if required.

6) The forward bioshield cap jettison (BCJ) preceeds VLC separation by a minimum time.

7) The VLC adapter jettison (AJ) shall occur as soon as possible after VLC separation and shall require a minimum maneuver.

4. Subsystem Assumptions

1) Science

a) Principal surveillance instruments located on articulating science platform.
b) Data modes and rates similar to Mariner Mars 1971.
c) High sensitivity VO imaging subsystem requiring no image motion compensation at periapsis.

2) Telecommunications

a) A DSN subnet of three 210' stations during orbital operations.
b) A DSN subnet of three 85' stations during cruise.
c) Mariner Mars 1971 telemetry and RF subsystems modified for new antennas and capsule requirements.
d) Telemetry modes shown in Table 7C-2.
e) Engineering telemetry during all maneuvers.
f) Command capability during all maneuvers.

3) Guidance and Control

a) NiCd secondary type battery allowing multiple maneuvers for orbit trim or planet surveillance and long cycle life.
b) Solar panel area and resulting charging times consistent with VO configuration concept.
c) CC&S capable of reprogramming.
d) CC&S modified to increased control of maneuvers, separations and other control functions.
e) CC&S requires 512 seconds (approximately 8 minutes) between turns and other control functions.
f) Maneuver times are computed based on a 670°/hour turn rate.
g) Maneuver and pointing capability of science platform sufficient for pre-separation surveillance events.

4) Structure (configuration)

a) Science platform axis on VO cone axis with 90° cone capability.
b) Stowable relay antenna array deployable to fixed position.
c) "Capsule up" launch configuration
d) Bioshield cap jettisoned in Mars orbit.
e) VLC adapter jettisoned before VLC atmospheric entry.

5) Propulsion

a) No direct VO control of bioshield cap and VLC separations.
b) Motor burns controlled by pyro subsystem in conjunction with CC&S.
c) Single propulsion engine requiring approximately 45 minutes burn time at orbit insertion.

5. Periapsis Passage Guidelines

1) Landing site surveillance on early passes (with lander on board) by maneuvering spacecraft to allow scan platform to slew in clock with a minimum cone angle change.
2) VL-VO communication in three subsequent passes after separation orbit.
3) Mapping of planet following sequence of VL communication passes.
4) Returning to within VL view for communications if required. (Nominally once per month).

Table 7C-1. Acronyms

A	VLC Adapter
A/C	Attitude Control Subsystem (Orbiter and Lander Capsule)
AMR	Altitude Marking Radar
AOS	Acquisition of Signal
BC	Bioshield Cap
CC&S	Orbiter Central Computer & Sequencer
CPS	Capsule Power System
CRS	Capsule Radio System
DSS	Data Storage Subsystem
DTR	Digital Tape Recorder
FCS	Flight Command Subsystem
FTS	Flight Telemetry Subsystem
LCE	Launch Complex Equipment
LC&S	Lander Computer & Sequencer
LOS	Loss of Signal
OPS	Operations
PAS	Pyrotechnic Arming Switch
Pwr	Power Subsystem
Pyro	Pyrotechnics Subsystem
RFS	Radio Frequency Subsystem
RRS	Relay Radio Subsystem
RTS	Relay Telemetry Subsystem
SIT	Separation Initiated Timer
TLM	Telemetry
VL	Viking Lander
VLC	Viking Lander Capsule
VO	Viking Orbiter
VS/C	Viking Spacecraft

Table 7C-2. Telemetry Modes Viking Orbiter Mission

Mode No.	Description	Time in Operation
1	VLC and/or VO Engineering (8 1/3 or 33 1/3 bps)	Cruise and Orbiter Maneuvers
2	Mode 1 (8 1/3 bps) +1348 bps down link	VLC operations with VO high-gain antenna toward Earth
	a) Hard line feedthrough (1348 bps)	VLC Checkout
	b) RF link feedthrough (1348 bps)	VLC entry
3	Mode 1 (8 1/3 bps) +1348 bps tape storage	VLC operations with VO high-gain antenna away from Earth
	a) Hard line input to DSS (1348 bps)	Preseparation operations
	b) RF link input to DSS (1348 bps)	Post-separation operations
4	Mode 1 (8 1/3 bps) + RF link from VL to DSS (20 Kbps)	VL relay passage.
5	Mode 1 (8 1/3 bps) + VO science to DSS (132 Kbps)*	VO science passage.
6	Mode 1 (8 1/3 bps) + high rate down link	No VO and VL acquisition operations. DSN 210' dish in view.
	a) Playback of recorded 1348 bps (16, 8, 4, 2, 1 Kbps)*	
	b) Playback of recorded 20 Kbps (16, 8, 4, 2, 1 Kbps)*	
	c) Playback orbiter science (16, 8, 4, 2, 1 Kbps)*	
7	Mode 1 (8 1/3 bps) + VO low rate science (133 1/3 bps)*	Science backup mode.

* Mariner Mars 1971 values (approximate).

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Table 7C-3. Orbiter Mission Sequence of Events
Chronological Index of Key Events

Event	Event Abbr.	Event Time	Time re P_s ($P_s \triangleq$ below)	Page No.
Countdown Begin		L-4h	--	7C-8
Launch	L	L	--	7C-8
Separation	S	L+41m	--	7C-9
Complete Attitude Stabilization		L+4h	--	7C-10
Start Capsule Monitor		L+8h	--	7C-10
Midcourse Maneuver Begin	M	L+5-15d	--	7C-11
Pre-encounter VO Updates		P_1-5d		7C-14
VO Insertion Burn Ignition	IB	P_1-23m		7C-17
Periapsis One	P_1	P_1		7C-17
1st Trim Burn	TB	P_2		7C-20
VO Preseparation Surveillance (typical)		P_3-20m	P_{s-10}-20m (nominal)	7C-23
VLC Checkout and Evaluation		P_{s-2}+1h		7C-25
Begin Separation Orbit		P_{s-1}		7C-26
Bioshield Cap Jettison	BCJ	P_s-3.7h		7C-27
VLC Separation	CS	P_s-2.8h		7C-29
VLC Adapter Jettison	AJ	P_s-1.7h	P_s-1.7h	7C-30
VLC Entry	E	E	P_s-17m	7C-31
VL Touchdown	TD	E+7m	P_s-11m	7C-31
Periapsis following Separation	P_s	P_s	P_s	7C-32
VL Relay Passage (typical)		P_{s+1}-25m	P_{s+1}-25m (nominal)	7C-33
VO Science Passage (typical)			P_{s+5}-20m (nominal)	7C-34
Period Trim Burn (typical)	TB		P_{s+6}-0.3m (nominal)	7C-37

PRELIMINARY

VIKING
SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
Launch Phase--Countdown Period					
1. Start final S/C countdown checkout		L-4h	LCE	All S/C Subsystems	A cursory verification of S/C launch readiness.
2. Clear, load & verify VO & VLC sequence		L-2h	LCE	CC&S, LC&S	
3. Switch S/C to internal pwr & verify		L-7m	LCE	Power	Verification monitored by LCE. TLM Mode 1 on
4. VLC power on		L-6m	CC&S, Power	CPS	Launch power mode
5. Reverify VO CC&S seq		L-6m, 30s	LCE	CC&S	
6. Start CC&S sequence		L-4m	LCE	CC&S	
7. Observe CC&S clear indication		L-1m	CC&S	LCE	
8. Stage 0 Ignition					
9. Liftoff		L			L=liftoff (2 in. rise) (nominal staging times)
10. Stage I start		L+1m,4[illegible]s			
11. Stage 0/I separation		L+2m,1[illegible]			
12. Stage I shutdown Stage II start		L+4m [illegible]s			
13. Jettison nose fairing		L+4m,[illegible]			
14. Stage II shutdown		L+7m,[illegible]s			
15. Stage III start		L+7m,[illegible]s			
16. Stage III shutdown		L+8m, 10s			

PRELIMINARY

VIKING
SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
17. Stage III 2nd burn		Mission Dependent			(E16+25m) nominal
18. Stage III 2nd shutdown		Mission Dependent			(E17+2 to 4m) nominal
19. <u>Separation</u>		(L+41m nom.) S=0	State III Programmer	Pyro	Stage III pyro unit provides circuit for release device S=S/C separation
a. Enable CC&S output			Cabling (Sep Conn)	CC&S	CC&S outputs are inhibited until after separation.
b. Arm pyro S/S			Devices (PAS)	Pyro	
c. Turn on A/C S/S			Devices (PAS)	A/C	Sun search initiated. Canopus sensor <u>on.</u> Backup CC&S
d. Deploy S/P			Devices		Passive
20. Enable SIT		S+10 to 100s	SIT	Pyro	Backup PAS
a. Arm pyro					
21. Deploy high gain antenna		E20+120s	Devices (SIT)	Pyro	Backup PAS
22. Deploy low gain antenna		E21+120s	CC&S	Pyro	
a. Deploy relay antenna			CC&S	Pyro	
23. Deploy solar panels		L+56m	CC&S	Pyro	Backup SIT
24. A/C S/S on		L+64m	CC&S	A/C	Backup PAS
25. Complete sun acq.		E24+0 to 30m			
a. Start roll search for Canopus.			A/C	A/C	Star map is telemetered.

VIKING
PRELIMINARY SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
26. Step lower Canopus brightness gate		L=0+N hr (N=1,2,3.)	CC&S	A/C	Cyclic CC&S command each hr throughout the mission, which, unless inhibited by A/C lowers the level of the lower Canopus brightness gate.
27. Complete Canopus acquisition		E24+1 to 4h			
28. VLC cruise power on		L+8h	CC&S, Power	CPS	
29. VLC monitor power on		L+9h	CC&S, Power	CPS	
	a. Start VLC monitor.		CC&S	LC&S (CRS)	VLC TLM rate 1050 bph for 1 hour (periodically)
Cruise Phase					
30. Cyclic for testing RFS		L+24N hr (N=1,2,3...)	CC&S	RFS	CC&S cyclic every 24 hours. Switches exciter or power amplifier in the event of RFS failure.
31. Ranging on.		E30+10m	FCS	RFS	Ranging to threshold
Maneuver Phase					
32. Transmit pitch turn duration, roll turn duration(s), total burn duration, scale factor and update motor gimbal angles.		L+5-15 days	FCS	CC&S	CC-1, CC-2, CC-4 & others

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PRELIMINARY

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SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
33. Propulsion enable		M-5h	FCS	Pwr, Pyro	Autopilot and gimbal power on and off.
34. Purge manifold		M-4h	CC&S	Pyro	Propulsion solenoid on and off. First burn only.
35. Pressurize propulsion & open propellant lines.		M-3h	CC&S	Pyro	
36. Maneuver initiate		M=0	FCS	A/C	M=midcourse maneuver sequence begins
a. Gyros on			CC&S	A/C	
b. Autopilot on			CC&S	A/C	
c. Accelerometers on			CC&S	A/C	
d. Gimbal actuators on			CC&S	A/C	
37. Switch roll to inertial reference		M+68m	CC&S	A/C	Canopus sensor is switched out of control loop.
a. Set roll polarity			CC&S	A/C	
b. Start roll turn			CC&S	A/C	
38. Stop roll turn		E37+D1	CC&S	A/C	$1s \leq D1 \leq 34m, 7s$
a. Reset roll polarity			CC&S	A/C	
b. Switch pitch & yaw to inertial ref.			CC&S	A/C	Sun sensor is switched out of control loop.

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SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
39. Set pitch polarity		E38+8m	CC&S	A/C	CC&S requires 512 seconds, (approx. 8 min) between turns
a. Start pitch turn			CC&S	A/C	
40. Switch to maneuver antenna if required.		After E39	FCS	RFS	
41. Stop pitch turn		E39+D2	CC&S	A/C	1s ≤ D2 ≤ 34m, 7s
a. Reset pitch polarity			CC&S	A/C	
42. Switch roll to inertial reference		E41+8m	CC&S	A/C	If E40 is required and RF signal is low.
a. Set roll polarity			CC&S	A/C	
b. Start roll turn			CC&S	A/C	
43. Stop roll turn		E42+D3	CC&S	A/C	1s ≤ D3 ≤ 34m, 7s
a. Reset roll polarity			CC&S	A/C	
44. Start burn		E43+8m	CC&S	Pyro	Propulsion solenoid on.
45. Stop burn		E44+0.4 to 20s	CC&S	Pyro	Accelerometer initiated.
46. Set roll polarity		E45+8m	CC&S	A/C	If E42, E43 are required, begin unwind maneuver.
a. Start roll turn			CC&S	A/C	
47. Stop roll turn		E46+D3	CC&S	A/C	D3 defined in event 43
a. Reset roll polarity			CC&S	A/C	

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SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
48. Set pitch polarity		E47+8m	CC&S	A/C	
a. Start pitch turn			CC&S	A/C	
49. Switch back to low gain antenna		After E48	FCS	RFS	Only if item 40 was required.
50. Stop pitch turn		E48+D2	CC&S	A/C	D2 defined in event 41.
a. Reset pitch polarity			CC&S	A/C	
51. Set roll polarity		E50+8m	CC&S	A/C	
a. Start roll turn			CC&S	A/C	
52. Stop roll turn		E51+D1	CC&S	A/C	D1 defined in event 38.
a. Reset roll polarity			CC&S	A/C	
53. Complete sun acq.		E52+8m	CC&S	A/C	
54. Complete Canopus acquisition.		E53+8m	CC&S	A/C	
a. Gyros off			CC&S	A/C	
55. Maneuver inhibit		After E54	CC&S	A/C	Resets fixed sequences. Places maneuver logic in non-tandem mode.
a. Inhibit CC&S			FCS	CC&S	Inhibits all CC&S maneuver-related output commands to A/C.
b. Inhibit solenoid			FCS	Pwr, Pyro	
56. Isolate pressurant		M+? days	FCS	Pyro	Optional--depends on maneuver sequencing and verification.
a. Close propellant lines			FCS	Pyro	

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SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
Cruise Phase					
57. Set Canopus cone angle No. 1		L+14 days	CC&S	A/C	Backup FCS
58. Set Canopus cone angle No. 2		L+38 days	CC&S	A/C	Backup FCS
59. Set Canopus cone angle No. 3		L+63 days	CC&S	A/C	Backup FCS
60. Set Canopus cone angle No. 4		L+83 days	CC&S	A/C	Backup FCS
61. Switch to high gain antenna if required.		P_1-40 days	FCS	RFS	
Second Maneuver Phase					
62. 2nd midcourse maneuver		P_1-30 days (nominal)	FCS	CC&S	Repeat events 32 through 54 if required, except E34. Propulsion subsystem remains unlocked thru orbit insertion and initial trim maneuvers.
Pre-Encounter Updates					
63. Prepare CC&S updates		P_1-5d, 4h	OPS	DSIF	
64. Perform OD		P_1-5d, 3.5h	OPS		
65. Update VO CC&S		P_1-5d	OPS/DSIF	FCS/CC&S	Read, write and verify.
66. Perform OD		P_1-2d	OPS		
67. Perform OD.		P_1-1d	OPS		
68. Perform OD		P_1-12h	OPS		

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SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
69. Perform OD and maneuver computation		P_1-9h 30m	OPS		
70. Prepare command sequence		P_1-6h	OPS	DSIF	
71. Update maneuver turn durations		P_1-4h 10m	OPS/DSIF	FCS/AC	
Orbit Insertion Phase					
72. Receive pitch turn duration, roll turn duration, burn duration scale factor, and update motor gimbal angles.		P_1-4h	FCS	CC&S	CC-1, CC-2, CC-4, and others
73. Propulsion enable		P_1-4h	FCS	Pwr, Pyro	Arm propulsion solenoid. Propulsion subsystem remains unlocked from second mid-course maneuver.
74. Perform OD		P_1-3h 35m	OPS		Data from P-8h to P3h30m
75. Maneuver initiate		1B-2h,48m	CC&S	A/C	(P_1-3h,11m)
a. Gyros & electronics on			CC&S	A/C	
b. Autopilot on			CC&S	A/C	
c. Gimbal actuators on			CC&S	A/C	Verify gimbal angles
d. Accelerometers on			CC&S	A/C	
76. Complete OD		P_1-2h 50m	OPS		

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SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
77. Perform maneuver computation		P_1-2h 40m	OPS		For burn maneuver updates
78. Switch roll to inertial reference		IB-1h,40m	CC&S	A/C	Canopus sensor is switched out of control loop. Roll 21°.
a. Switch to low gain antenna			CC&S	RFS	
b. Gimbal actuators off			CC&S	A/C	
c. Set roll polarity			CC&S	A/C	
d. Start roll turn			CC&S	A/C	
79. Stop roll turn		IB-1h,38m	CC&S	A/C	(nominal time)
a. Reset roll polarity.			CC&S	A/C	
b. Switch pitch & yaw to inertial reference.			CC&S	A/C	Sun sensor is switched out of control loop.
80. Complete maneuver computation		P_1-1h 55m	OPS		
81. Set pitch polarity		IB-1h,30m	CC&S	A/C	Pitch off sun (P_1-1h,53m)
a. Start pitch turn			CC&S	A/C	Pitch 134°.
82. Switch to maneuver antenna		After E81	CC&S	RFS	If required.
83. Generate and verify burn update commands		P_1-1h50m	OPS	DSIF	

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SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
84. Stop pitch turn		IB-1h,18m	CC&S	A/C	
a. Reset pitch polarity			CC&S	A/C	
85. Set roll polarity		IB-1h,10m	CC&S	A/C	Roll turn required for communications.
a. Start roll turn			CC&S	A/C	
86. Stop roll turn		IB-1h,8m	CC&S	A/C	(nominal time)
a. Reset roll polarity			CC&S	A/C	
87. Transmit updates		IB-40m	OPS/DSIF	FCS/CC&S	
88. Receive update burn duration		IB-30m	FCS	CC&S	(P_1-53m) Inhibit command possible. Allow 30m for corrective action before burn.
a. Gimbal actuators on.			CC&S	A/C	
89. Start burn		IB	CC&S	Pyro	P_1-23m,prop. solenoid on
90. Periapsis		P_1			P_1=0=1st periapsis (L+220 days nominal)
91. Stop burn		P_1+23m	CC&S	Pyro	CC&S monitors acceleration to terminate burn.
a. Autopilot off			CC&S	A/C	Propulsion solenoid off.
b. Gimbal actuators off			CC&S	A/C	
92. Set roll polarity		P_1+27m	CC&S	A/C	Begin unwind.
a. Start roll turn			CC&S	A/C	

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SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
93. Stop roll turn		P_1+29m	CC&S	A/C	(nominal time)
a. Reset roll polarity			CC&S	A/C	
94. Set pitch polarity		P_1+37m	CC&S	A/C	
a. Start pitch turn			CC&S	A/C	Reacquire Sun.
95. Switch back to low gain antenna		After E94	CC&S	RFS	Only if E82 was required.
96. Stop pitch turn		P_1+49m	CC&S	A/C	Reacquire Sun, begin battery charging.
a. Reset pitch polarity			CC&S	A/C	
97. Set roll polarity		P_1+57m	CC&S	A/C	
a. Start 2nd roll turn			CC&S	A/C	
98. Stop roll turn		P_1+59m	CC&S	A/C	(nominal time)
a. Reset roll polarity			CC&S	A/C	Orbit insertion maneuver complete.
99. Propulsion inhibit		P_1+61m	FCS	Pwr, Pyro	Disarm propulsion solenoid. CC&S Backup.
100. Switch to celestial references		P_1+69m	CC&S	A/C	
101. Begin OD tracking		P_1+70m	OPS		
102. OD complete		P_1+15h55m	OPS		Data from P_1+1 to P_1+14h

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SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
103. Maneuver computation complete		P_1+17h 10m	OPS		
104. Generate and verify Maneuver Command Sequence		P_1+17h 20m	OPS	DSIF	
105. Transmit maneuver parameters		P_1+18h,10m	OPS/DSIF	FCS/CC&S	(P_2-6h 37m nominal)
Trim Maneuver					
106. Receive maneuver parameters		P_2-6h 17m	FCS	CC&S	See event 72.
a. Propulsion enable			FCS	Pwr, Pyro	Arm propulsion solenoid.
107. Trim maneuver initiate		TB-N_1			Trim burn maneuver begins.
a. Gyros & electronics on			CC&S	A/C	$N_1 \leq$ 194m (P_2-N_1)
b. Gimbal actuators on			CC&S	A/C	
c. Accelerometers on			CC&S	A/C	
d. Autopilot on			CC&S	A/C	
108. Switch roll to inertial reference.		TB-N_2	CC&S	A/C	Canopus sensor is switched $N_2 \leq$ 126m
a. Set roll polarity			CC&S	A/C	
b. Start roll turn			CC&S	A/C	

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SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
109. Stop roll turn		E108+D1	CC&S	A/C	$1s \leq D1 \leq 34m$
a. Reset roll polarity			CC&S	A/C	
b. Switch pitch and yaw to inertial reference			CC&S	A/C	Sun Sensor is switched out of control loop.
110. Set pitch polarity		E109+8m	CC&S	A/C	$(TB-N_3)$, $N_3 \leq 82m$
a. Start pitch turn			CC&S	A/C	Pitch off Sun
111. Switch maneuver antenna if required.		After E110	CC&S	RFS	
112. Stop pitch turn		E110+D2	CC&S	A/C	$1s \leq D2 \leq 34m$
a. Reset pitch polarity			CC&S	A/C	
113. Set roll polarity		E112+8m	CC&S	A/C	If E111 is required and RF signal margin is low.
a. Start roll turn			CC&S	A/C	
114. Stop roll turn		E113+D3	CC&S	A/C	$1s \leq D3 \leq 34m$ $(TB-N_4)$ $N_4 \leq 8m$
a. Reset roll polarity			CC&S	A/C	
115. Start trim burn		TB	CC&S	Pyro	(P2), Prop. solenoid on
116. Stop burn		E115 +0.4 to 20s	CC&S	Pyro	$(TB+K_1)$, $K_1 \leq 1m$, prop. solenoid off.
a. Autopilot off			CC&S	A/C	
b. Gimbal actuators off			CC&S	A/C	

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SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
117. Set roll polarity		E116 +8m	CC&S	A/C	Begin unwind maneuver.
a. Start roll turn			CC&S	A/C	
118. Stop roll turn		E117 +D3	CC&S	A/C	D3 defined in event 114 (TB+K_2), $K_2 \leq 43$m
a. Reset roll polarity			CC&S	A/C	
119. Set pitch polarity		E118 +8m	CC&S	A/C	
a. Start pitch turn			CC&S	A/C	
120. Switch back to low gain antenna		After E119	CC&S	RFS	If event 111 was required.
121. Stop pitch turn		E120 +D2	CC&S	A/C	D2 defined in event 112 (TB+K_3), $K_3 \leq 85$m
a. Set roll polarity			CC&S	A/C	Reacquire Sun (P2+K3), battery charger on.
122. Stop roll turn		E121+D1	CC&S	A/C	D1 defined in event 109 (TB+K4) K4 $\leq$ 119m (P2+K4)
a. Reset roll polarity			CC&S	A/C	Trim maneuver complete
b. Switch to H/G ant.			CC&S	RFS	
123. Propulsion inhibit		P_2+2h7m	FCS	Pwr, Pyro	Disarm propulsion solenoid.
124. Begin orbit determination			OPS		
125. OD complete		P_2+16hr	OPS		
126. Repeat events 103 thru 125.					If required for orbit trimming

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SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
127. Maneuver computation complete		P2+17h15m	OPS		
a. Update CC&S			FCS	CC&S	If required for trimmed orbit
128. Unlatch scan platform		P3-3h	CC&S	Pyro	
VO Preseparation Surveillance Phase					
128. Maneuver initiate		P3-2h19m			
a. Gyros on			CC&S	A/C	
129. Science power on		P3-71m	CC&S, power	Science	
130. Switch to L/G ant.			CC&S	RFS	
131. Switch roll to inertial reference		P_3-64m	CC&S	A/C	Begin mapping maneuver.
a. Set roll polarity			CC&S	A/C	
b. Start roll turn			CC&S	A/C	
132. Stop roll turn		P_3-57m	CC&S	A/C	
a. Reset roll polarity			CC&S	A/C	
b. Switch pitch & yaw to inertial reference			CC&S	A/C	
133. Set pitch polarity		P_3-49m	CC&S	A/C	Pitch off Sun
a. Start pitch turn			CC&S		

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SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
134. Stop pitch turn		P_3-43m	CC&S	A/C	
a. Reset pitch polarity					
135. Set roll polarity		P_3-35m	CC&S	A/C	To point within clock angle capability of scan platform.
a. Start roll turn			CC&S	A/C	
136. Stop roll turn		P_3-28m	CC&S	A/C	Inertial hold.
a. Reset roll polarity			CC&S	A/C	
137. Begin Surveillance		P_3-20m	CC&S	Science	
a. Scan sequence begin			CC&S	A/C	Slew scan platform in clock with minimum cone change.
b. Switch to TLM mode 5			CC&S	FTS	
138. End Surveillance		P_3+5m	CC&S	Science	
a. Scan science off			CC&S	Science	
b. Switch to TLMmode 1			CC&S	FTS	
139. Set roll polarity		P_3+13m	CC&S	A/C	Begin unwind maneuver
a. Start roll turn			CC&S	A/C	
140. Stop roll turn		P_3+20m	CC&S	A/C	
a. Reset roll polarity			CC&S	A/C	
141. Set pitch polarity		P_3+26m	CC&S	A/C	
a. Start pitch turn			CC&S	A/C	

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SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
142. Stop pitch turn		P_3+34M	CC&S	A/C	Reacquire Sun, battery charger on.
a. Reset pitch polarity			CC&S	A/C	
143. Set roll polarity		P_3+42m	CC&S	A/C	
a. Start roll turn			CC&S	A/C	
144. Stop roll turn		P_3+49m	CC&S	A/C	
a. Reset roll polarity			CC&S	A/C	
145. Switch to celestial references		P_3+57m	CC&S	A/C	Surveillance maneuver complete
a. Gyros off			CC&S	A/C	
146. Switch to H/G antenna			CC&S	RFS	
147. Switch to TLM mode 6c.		P_3+66m	CC&S	FTS	Playback Surveillance data
a. DSS on			CC&S	DSS	
148. Switch to TLM mode 1.		P_3+8h	CC&S	FTS	End playback.
149. Repeat events 128 thru 148		P_n			nth orbit, where $4 \leq n \leq 30$, if required.

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SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
Period Trim					To provide a Mars synchronous orbit.
150. Complete OD.		P_{s-3}-9h	OPS		Data from P_{s-4}-1h to P_{s-4}+14h
151. Maneuver computation complete.		P_{s-3}-7h27m	OPS		
152. Generate and verify maneuver command sequence.		P_{s-3}-7h17m	OPS	DSIF	
153. Transmit maneuver parameters		P_{s-3}-6h27m	OPS/DSIF	FCS/CC&S	
154. Repeat events 106 thru 125					
155. Repeat events 150 thru 154					If required for orbit tuning.
VLC Checkout Orbit					
156. Start VLC checkout		P_8+1h			P_8 is nominal periapsis number (P_{s-2})
a. Switch to TLM mode 2a.			CC&S	FTS	Hard line feed through (1348 bps)
b. VLC checkout power on			CC&S, Power	CPS	
	157. VLC checkout begin	P_8+1h	CC&S	LC&S	
158. End VLC checkout		P_8+4h			Followed by up to 12 hours of assessment and evaluation

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SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
a. Switch to TLM mode 1			CC&S	FTS	
b. VLC checkout, power off			CC&S	Power	
	159. VLC checkout end	P_8+4h	CC&S	LC&S	
VLC Separation Orbit					
160. Complete OD		P_s-9h	OPS		Data from P_{s-1}+1h to P_{s-1}+14h
161. Maneuver computation complete		P_s-7h27m	OPS		
162. Generate and verify maneuver command sequence		P_s-7h17m	OPS	DSIF	
163. Transmit maneuver parameters		P_s-6h27m	OPS/DSIF	FCS/CC&S	
164. Receive maneuver parameters		P_s-6h17m	FCS	CC&S	Read, write, and verify.
Bioshield Cap Jettison Maneuver					
165. Maneuver initiate		BCJ-84m			(P_s-5h,6m) (nominally P_{11})
a. Switch to low gain antenna			CC&S	RFS	
b. Gyros on			CC&S	A/C	

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VIKING
SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
166. Switch roll, pitch & yaw to inertial references		BCJ-16m	CC&S	A/C	Sun sensor out of control loop.
a. Set pitch polarity			CC&S	A/C	
b. Start pitch turn			CC&S	A/C	Pitch off Sun
167. Switch to maneuver antenna if required		After E166	CC&S	RFS	
168. Stop pitch turn		BCJ-8m	CC&S	A/C	
a. Reset pitch polarity			CC&S	A/C	
169. Bioshield cap jettison		BCJ	LC&S	Pyro	BCJ=0=Bioshield Cap jettison (P_s-3h,42m)
170. Set pitch polarity		BCJ+8m	CC&S	A/C	Begin unwind
a. Start pitch turn			CC&S	A/C	
171. Switch back to low gain antenna		After E134	CC&S	RFS	Only if E131 was required.
172. Stop pitch turn polarity		BCJ+16m	CC&S	A/C	Reacquire Sun Jettison maneuver complete (P11-3h,26m)
a. Reset pitch polarity			CC&S	A/C	
<u>VLC Separation Maneuver</u>					
173. Switch to high gain antenna		CS-32m	CC&S	RFS	(P_s-3h 18m nominal)

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SEQUENCE OF EVENTS

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ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
a. Switch to TLM mode 2a.			CC&S	FTS	Hard line feed through 1348 bps
	b. VLC Xmitter on		LC&S	CRS	
174. Switch to low gain antenna		CS-24m	CC&S	RFS	
a. Switch to TLM mode 3a			CC&S	RTS	Record hard line (1348 bps)
	b. VLC to internal power		CC&S/PWR	LC&S/CPS	
175. Switch roll to inertial reference		CS-24m	CC&S	A/C	
a. Set roll polarity			CC&S	A/C	
b. Start roll turn			CC&S	A/C	
176. Stop roll turn		CS-20m	CC&S	A/C	
a. Reset roll polarity			CC&S	A/C	
177. Switch pitch & yaw to inertial reference		CS-20m	CC&S	A/C	Sun sensor out of control loop.
178. Set pitch polarity		CS 12m	CC&S	A/C	Pitch off Sun(P_s-2h,58m)
a. Start pitch turn			CC&S	A/C	
179. Stop pitch turn		CS-8m	CC&S	A/C	
a.Reset pitch polarity			CC&S	A/C	

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ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
180. VLC separation		CS	LC&S	Pyro	CS=0=VLC separation (P_s-2h,46m)
a. Switch to Antenna L			CC&S	RRS	Antenna L provided by VLC within bioshield base.
b. Switch to TLM mode 3b			CC&S	RTS	Record 1348 bps.
	182. RF link established	CS+0.5s	LRS	RRS	
	183. Ignite deflection propulsion	CS+30m	LC&S	VL Propulsion	(P_s-2h,16m)
	184. Terminate deflection propulsion	CS+30m, 12s	LC&S	VL Propulsion	
185. Switch to TLM mode 1		P_s-2h 15m	CC&S	FTS	
	a. Lander transmitter off	CS+31m	LC&S	CRS	
186. Set pitch polarity		CS+38m	CC&S	A/C	Begin unwind maneuver.
a. Start pitch turn			CC&S	A/C	(P_{11}-2h8m)
187. Stop pitch turn		CS+42m	CC&S	A/C	Reacquire Sun, battery charger on (P_s-2h,4m)
a. Reset pitch polarity			CC&S	A/C	
188. Set roll polarity		CS+50m	CC&S	A/C	
a. Start roll turn			CC&S	A/C	

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SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
189. Stop roll turn		CS+54m	CC&S	A/C	
a. Reset roll polarity			CC&S	A/C	Separation maneuver complete (P_s-1h,52m)
b. Gyros off			CC&S	A/C	
190. Switch to H/G antenna		CS+54m	CC&S	RFS	
<u>Adapter Jettison Event</u>					No maneuver
191. Initiate Event		AJ-8m	CC&S	A/C	Sun Celestial sensor in control (P_s-1h,44m)
192. VLC adapter jettison		AJ	CC&S	Pyro	(P_s-1h,36m) AJ=0=VLC adapter (sun line) jettison
193. Terminate event		AJ+8m	CC&S	A/C	Adapter jettison event complete (P_s-1h,28m)
<u>Entry Phase</u>					
194. Switch to TLM mode 2b		P_s-1h	CC&S	FTS	Relay 1348 bps direct
a. Switch to relay antenna			CC&S	RRS	
b. Establish RF link with VLC			RFS	CRS	
	c. VLC Xmitter on		LC&S	CRS	

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ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
	195. Jettison thrust cone	E-11m,45s	LC&S	Pyro	Entry & landed events & time have been assumed except where noted. (Updated info is expected from LaRC.)
	196. Entry Science on	E-11m	LC&S	Science	
	a. Altitude marking radar (AMR) on		LC&S	Science	
	197. Entry	$E(P_s\text{-}17m)$			E=0=Entry Altitude = $8x10^5$ ft $\gamma_E = -16^o$ $V_E = 1.53 \times 10^4$ ft/sec Assume LaRC maximum atmosphere per LaRC
	198. Mach 5 (initiate Entry science)	E+1m,20s	Sensor	Science	
	199. Deploy chute		Sensor/AMR	Pyro	Altitude 21000 ft
Landing Phase					
	200. Separate aeroshell		LC&S	Pyro	
	a. Retro engines on		LC&S	VL Propulsion	
	201. Jettison parachute	E+5m	AMR/LC&S	Pyro	AMR 100 ft mark
	202. Touchdown	TD (E+7m)			TD=0=Touchdown (P_s -11m)
	a. VL to full surface power		LC&S	CRS	
203. Switch to TLM mode 4			CC&S	RTS	

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SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
	a. VL Xmitter on	P_s -10m	LC&S	CRS	20 kbps readout from surface
	204. Deploy surface sci.	TD+2m	LC&S	Pyro	
	205. Initiate surface sci. (imagery and meteorology)	TD+2m	LC&S	Science	2 meteorology samples required, repeat 8m, 30s later.
206. Periapsis		P_s			P_s=0=11th periapsis (nominal)
207. Switch to TLM mode 1		P_s+5m	CC&S	FTS	
	a. VLC Xmitter off		LC&S	CRS	
208. Switch to TLM mode 6a.		P_s +6m	CC&S	FTS	Begin preentry data playback.
a. DSS on			CC&S	DSS	
209. Switch to TLM mode 6b.		P_s+15m	CC&S	FTS	Begin landed data playback.
a. End entry playback.			CC&S	DSS	
210. Switch to TLM mode 1		P_s+52m	CC&S	FTS	End playback
a. DSS off			CC&S	DSS	

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ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
VL Relay Passage					
211. Beacon Xmitter on		P_{12}-25m	CC&S	RRS	Nominal periapsis number. (P_{s+1}-25m)
a. Switch to relay antenna			CC&S	RRS	
	212. Beacon receiver on	P_{12}-20m	LC&S	CRS	
	213. Beacon AOS	P_{12}-18m	CRS		
	a. Surface science measurements off		LC&S	Science	
	b. Memory storage mode off		LC&S	DSS	
214. VO RF link established		P_{12}-17m	RFS	CRS	
a. Beacon Xmitter off			CC&S	RRS	Acquire and record VL data at 20 kbps.
b. Switch to TLM mode 4			CC&S	RRS, DSS	
	215. Beacon receiver off	P_{12}-17m			(nominal time)
	a. High TLM rate on		LC&S	CRS	
216. Periapsis		P_{12}			P_{12}=0=12th periapsis (nominal)
	217. High TLM rate off	P_{12}+4m	LC&S	RRS, DSS	VL readout complete
218. VO-VL LOS		P_{12}+5m			
a. Switch to TLM mode 1.			CC&S	FTS	
	b. VLC Xmitter off		LC&S	CRS	

PRELIMINARY

VIKING
SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
219. Switch to TLM mode 6b		P_{12}=10m	CC&S	FTS, DSS	Playback stored VL data
22C. Switch to TLM mode 1		P_{12}+60m	CC&S	FTS, DSS	Complete playback.
221. Repeat events 211 thru 220		P_n			3 out of 4 of the following passes (nominally). May be changed if VL direct link fails.
VO Science Acquisition Passage					
222. Science Power on		P-60m	CC&S	Power	Begin mapping, w/o capsule. (P=typical periapsis following VLC communication periods)
a. Slew platform to 1st TV position			CC&S	Science	
b. TV & sci.warmup			Power	Science	
223. Begin mapping		P-20m			Altitude 2500 km.
a. Start recorder			CC&S	DSS	
b. Slew platform in cone and clock.			CC&S	Science	Scan platform corrected to local vertical for each frame.
224. End mapping		P+3m			
a. Stop recorder			CC&S	DSS	
b. Platform pwr off			CC&S	Science	
225. Switch to TLM mode 6c.		P+11m	CC&S	FTS	Begin playback
a. DSS on			CC&S	DSS	

VIKING
PRELIMINARY SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
226. Switch to TLM mode l		P+7h			End playback
a. DSS off					
227. Repeat events 221 thru 226					Until sufficient map of lander area is acquired.
Period Trim Maneuver					To allow broad planet surveillance following VL communications periods
228. Complete OD		P-9h	OPS		Data from 1h to 14h from previous periapsis
229. Maneuver computation complete		P-7h 27m	OPS		
230. Generate and verify maneuver command sequence		P-7h 17m	OPS	DSIF	
231. Transmit maneuver parameters		P-6h 27m	OPS/DSIF	FCS/CC&S	
232. Receive maneuver parameters		P-6h 17m	FCS	CC&S	Read, write, and verify
233. Trim maneuver initiate		P-N_1	CC&S	A/C	($N_1 \leq 194$m) Open propellant and pressurant lines if required.
a. Gyro electronics on			CC&S	A/C	
b. Gimbal actuators on			CC&S	A/C	
c. Accelerometers on			CC&S	A/C	

PRELIMINARY

VIKING
SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
d. Switch to low gain antenna #1			CC&S	RFS	
234. Switch roll to inertial reference		$P-N_2$	CC&S	A/C	Canopus sensor out of control loop ($N_2 \leq 126m$)
a. Autopilot on					
b. Set roll polarity					
c. Start roll turn					
235. Stop roll turn		E233+D1	CC&S	A/C	$1s \leq D1 \leq 34m$
a. Reset roll polarity			CC&S	A/C	
b. Switch pitch & yaw to inertial ref.			CC&S	A/C	
236. Set pitch polarity		E234+8m	CC&S	A/C	$(P-N_3)$ ($N_3 \leq 82m$)
a. Start pitch turn			CC&S	A/C	Pitch off Sun
237. Switch to maneuver antenna if required		After E235	CC&S	RFS	
238. Stop pitch turn		E235+D2	CC&S	A/C	$1s \leq D2 \leq 34m$
a.Reset pitch polarity			CC&S	A/C	
239. Set roll polarity		E237+8m	CC&S	A/C	
a.Start roll turn			CC&S	A/C	
240. Stop roll turn		E238+D3	CC&S	A/C	$1s \leq D3 \leq 34m$

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VIKING
SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
a.Reset roll polarity			CC&S	A/C	$(P-N_4)$ $N_4 \leq 8m$
b.Propulsion enable			FCS	Pwr, Pyro	Arm propulsion solenoid.
241. Start trim burn		TB	CC&S	Pyro	$P-N_5$ 2, $N_5 \leq 20s$, propulsion solenoid on.
242. Periapsis		P			P=0=periapsis (typically P_{16})
243. Stop burn		$E240+N_5$	CC&S	Pyro	Propulsion solenoid off.
a. Autopilot off			CC&S	A/C	$(P+K_1)$, $K_1 \leq 1m$
b. Gimbal actuators off			CC&S	A/C	
244. Set roll polarity		E242+8m	CC&S	A/C	Begin unwind.
a. Start roll turn			CC&S	A/C	
245. Stop roll turn		E243+D3	CC&S	A/C	D3 defined in E239
a.Reset roll polarity			CC&S	A/C	$(P+K_2)$, $K_2 \leq 43m$
246. Set pitch polarity		E244+8m	CC&S	A/C	
a.Start pitch turn			CC&S	A/C	
247. Switch back to low gain antenna		After E245	CC&S	RFS	If E236 was required.
248. Stop pitch turn		E245+D2	CC&S	A/C	D2 defined in E237 $(P+K_3)$, $K_3 \leq 85m$
a.Reset pitch polarity			CC&S	A/C	Reacquire Sun, begin battery charging

PRELIMINARY

VIKING
SEQUENCE OF EVENTS

ORBITER	LANDER	TIME	SOURCE	DESTINATION	COMMENT
249. Set roll polarity		E247+8m	CC&S	A/C	
a.Start roll turn			CC&S	A/C	
250. Stop roll turn		E248+D1	CC&S	A/C	D1 defined in E234 $(P+K_4)$, $K_4 \leq$ 2h,7m
a.Reset roll polarity			CC&S	A/C	
b.Acquire sun and Canopus			CC&S	A/C	
c. Gyros off			CC&S	A/C	
d. Propulsion inhibit			FCS	Pwr, Pyro	
e.Isolate pressurant			FCS	Pyro	
f.Close propellant lines			FCS	Pyro	
251. Repeat events 222 thru 226					VO Science acquisition Repeat until VL requires relay then go to event 252.
252. Repeat events 228 thru 250					Orbit trim to allow reacquisition of lander relay data or further mapping.
253. Repeat events 211 through 220					Relay passage following trim.
254. Repeat event 252					Trim to continue mapping.
255. Repeat event 251					For remainder of mission. (Mission & science strategy not defined as of 1/3/69.)

D. ORBITER WEIGHT

The VO design concept accommodates both orbiter science similar to MM71 and a relay telemetry and capsule support capability. The VO must support the capsule in orbit for an out-of-orbit delivery mode. The large weight penalties for this mode are well known.

The following subsections discuss the basis for the VO weights and a history of the iterations to this date.

1. Basis for the VO Weights

Weight estimates for the VO subsystems are based on the latest MM71 estimates available at this time. Relay subsystem weights are derived from JPL estimates.

Significant weight changes from MM71 have occurred for three subsystems: propulsion, structure, and power. Out-of-orbit VLC delivery has required a large change in the propulsion subsystem. The design concept weights are derived from the latest MM71 technology and include the new four-tank designs for propellant and pressurant. Major contributors to increased weight in the structural subsystem have been the new wrap-around bus and the four additional electronic chassis. The cyclic power modes of orbiter operation require a secondary type of battery in the power subsystem. A NiCd design was chosen to tolerate the large number of discharge cycles consistent with repetitive orbit maneuvers off the Sun line and possible periodic solar occultations late in the mission. The NiCd battery weight penalties coupled with the larger solar panel area, designed for the increased solar distances in 1974, are responsible for the large power subsystem changes. These will be discussed in Section VII. E.

2. Current Orbiter Weights

The current VO weights presented in Table 7D-1 represent a JPL subsystem weight iteration including the negotiated JPL-LaRC contingencies.

	SUBSYSTEM	CURRENT (2/17/69) WEIGHT(lb)	JPL-LaRC CONTINGENCY FACTOR
01	Structure	284.5	1.15
02	Radio Frequency	74.4	1.05
03	Command	11.7	1.05
04	Power	258.4	1.05
05	Central Computer and Sequencer	27.5	1.05
06	Flight Telemetry	26.0	1.10
07	Attitude Control	112.3	1.10
08	Pyrotechnics	15.0	1.10
09	Cabling	109.0	1.10
11	Temperature Control	27.0	1.15
12	Devices	72.6	1.15
16	Data Storage	40.0	1.10
20	Science Data	22.0	1.15
31	Scan Control	16.7	1.05
	Science Instruments	128.3	1.10
52	Relay Radio	18.0	1.20
56	Relay Telemetry	10.0	1.20
	VO Weight (less Propulsion Subsystem)	1253.0	
10	Propulsion	3650.0*	
	VO Injected Weight	4903.0	
	VLC Adapter Weight**	40.0	1.15
	VLC Weight	1800.0	
	VS/C Injected Weight	6743.0	
	VS/C Adapter Weight***	285.0	1.15
	VS/C Launch Weight	7028.0	

*Assumes same inerts without contingencies as 4105-lb case but off loaded to retain

**Weight included in VO weight for propulsion subsystem sizing.

***Includes destruct package, diaphragm, ring, but not encapsulation diaphragm bulkh

FOLDOUT FRAME /

:XPECTED WEIGHT(lb)	WEIGHT ASSUMPTIONS
327.0	Capsule up, new wrap-around bus, new fold-out solar arrays.
78.0	MM'71, with 58" high gain antenna.
12.5	MM'71
271.0	MM'71 panel technology, NiCd battery, new charger.
29.0	MM'71 (mod.)
28.5	MM'71 with mods for capsule and lander relay.
123.5	MM'71 (mod.)
16.5	MM'71 (mod.)
120.0	MM'71 (mod.)
31.0	MM'71 (mod.)
83.5	MM'71 (mod.)
44.0	MM'71, two digital tape recorders.
25.5	MM'71 mod for new science.
17.5	MM'71
141.0	New baseline payload.
21.5	New
12.0	New
1382.0	
3980.0	New four tank design, MM'71 engine and related hardware ΔV = 1350 m/sec (impulsive orbit insertion)
5362.0	
48.0	
1800.0	
7210.0	
328.0	
7538.0	

ıaximum ΔV - 1350 m/sec (see Section IX. C.)

ɪd.

Table 7D-1. Viking Orbiter Subsystem Weight Summary

E. ORBITER POWER PROFILE

The orbiter power subsystem consists of a primary and a secondary power source, solar panels, and nickel cadmium batteries, respectively. In addition, power conditioning components are provided for the conditioning and conversion of unregulated solar panel and battery outputs as required for user subsystems. A functional block diagram of the orbiter baseline power subsystem is shown in Figure 8F-3. The Mariner '71 power subsystem was used as the baseline system. The sizing of the primary and secondary source and conditioning equipment is based on the total power demand required from each source and component in addition to the efficiencies of these components which vary throughout the mission. Component efficiences and how they affect the total power demand are shown in Table 8F-1a. Consideration is also given to the natural environment of the planet related to the sizing of the solar panel. Because of the increased distance of Mars with respect to Sun as compared to the Earth-Sun distance, the solar intensity is decreased by approximately 65% (90 days after arrival). As a consequence, solar panel available power is decreased from 1082 watts (60° C) at Earth to 464 watts (-16° C) at Mars. The solar panel area required to provide 464 watts at Mars is 104 ft^2. The sizing of the batteries has been based on a 167 minute worst case energy demand period. This period considered as the orbit insertion mode, requires 1195 watt hours. Hermetically sealed nickel cadmium batteries have been selected for this mission because of the capability to provide 70-75 flight cycles.

Table 7E-1 summarizes the total power required per the key operational modes which occur from launch through 20 Mar 74 Mars arrival and 90 days after arrival. The primary and secondary power source that would be available for providing the power is also included. Of significance in power demand are the Viking lander capsule checkout requirements of 182 watts for a period of approximately three hours, the TWT subsystem continuous requirement of 96 watts, and battery charging requirements of 125 watts. A detailed breakdown of power requirements is shown in Table 8F-1a.

Table 7E-1. Preliminary Power Profile (Baseline)

OPERATIONAL MODE	POWER REQUIRED (watts)	POWER SOURCE
1. LAUNCH	326	BATTERY
2. SUN ACQUISITION	338	BATTERY
3. CRUISE I BATTERY CHG ON (GYROS OFF)	434	SOLAR PANEL
4. CRUISE II BATTERY CHG OFF	296	SOLAR PANEL
5. CRUISE III V.L.C. CHECKOUT	366	SOLAR PANEL
6. MANEUVER	469, 357	BATTERY
7. CRUISE I BATTERY CHG ON	434	SOLAR PANEL
8. CRUISE II BATTERY CHG OFF	334	SOLAR PANEL
9. ENCOUNTER	445	SOLAR PANEL
10. GYRO WARM UP ON	378	SOLAR PANEL
11. ORBIT INSERTION MANEUVER	391	SOLAR PANEL
12. ORBIT INSERTION MANEUVER	403	BATTERY
13. ORBIT INSERTION MANEUVER BURN	518	BATTERY
14. ORBIT INSERTION MANEUVER AND UNWIND	389	BATTERY
15. ORBIT CRUISE WITH V.L.C.	463	SOLAR PANEL
16. ORBIT TRIM WITH V.L.C.	518①, 406②	BATTERY
17. ORBIT CRUISE WITH V.L.C.	463	SOLAR PANEL
18. ORBIT CRUISE WITH V.L.C. AND SCIENCE	515	BATTERY
19. ORBIT CRUISE WITH CAPSULE	463	SOLAR PANEL
20. V.L.C. PRESEPARATION CHECKOUT	507	SP/BATTERY
21. V.L.C. EVALUATION AND ASSESSMENT	473	SOLAR PANEL
22. GYRO WARM UP (BC JET MAN.)	390	SOLAR PANEL
23. BC JET MANEUVER	390	SOLAR PANEL
24. BC JET MANEUVER	401	BATTERY
25. V.L.C. SEPARATION MANEUVER	373	SOLAR PANEL
26. V.L.C. SEPARATION	387	BATTERY
27. V.L.C. SEPARATION MANEUVER UNWIND	387	SOLAR PANEL
28. ADAPTER JET MANEUVER	461	SOLAR PANEL
29. ENTRY RELAY	333	SOLAR PANEL
30. ORBITAL SCIENCE ON	430	SOLAR PANEL
31. ORBITAL SCIENCE PLAYBACK	330	SOLAR PANEL
32. ORBITAL CRUISE (SUN OCCULTATION)	360	BATTERY
33. ORBITAL CRUISE WITH BATTERY CHG	461	SOLAR PANEL

① 1 MIN ON; ② 59 MIN ON.

F. ORBITER DATA HANDLING

1. Introduction

The Viking orbiter data handling is more complex than Mariner '71 orbiter data handling because of the larger number of data modes required by the various phases of the Viking mission. Additional flexibility is also required to support the lander capsule during capsule checkout, entry, and the subsequent landing at which time the orbiter performs in the capacity of a relay satellite. Because of time criticality of the encounter and orbital mode changes, the modes shall be initiated by the CC&S program.

The seven orbiter data handling modes considered as a baseline to this conceptual design and utilized in the development of the flight sequence are delineated in the subsequent paragraphs of this section. The assumed constraints and data requirements are also outlined as the various data modes are individually developed.

2. Mode Descriptions

Mode 1 - This mode is used for low data rate monitoring of (1) spacecraft throughout interplanetary transit (includes midcourse and insertion maneuver) and of (2) orbiter after insertion during "quiet" time (includes monitoring of lander capsule prior to separation). "Quiet" time is defined as when (1) no capsule data is being received by the orbiter, and (2) when the orbiter is not performing scientific experiments, i.e., the only orbiter function is "housekeeping" requiring low data rates. This mode requires a single channel. The data rate is 8 1/3 or 33 1/3 bps. See Figure 7F-1.

Mode 2a - Engineering and hardline capsule checkout is being sent in real time since Earth is in view of high-gain antenna. This mode requires two channels. Engineering channel is 8 1/3 bps and hardline capsule is

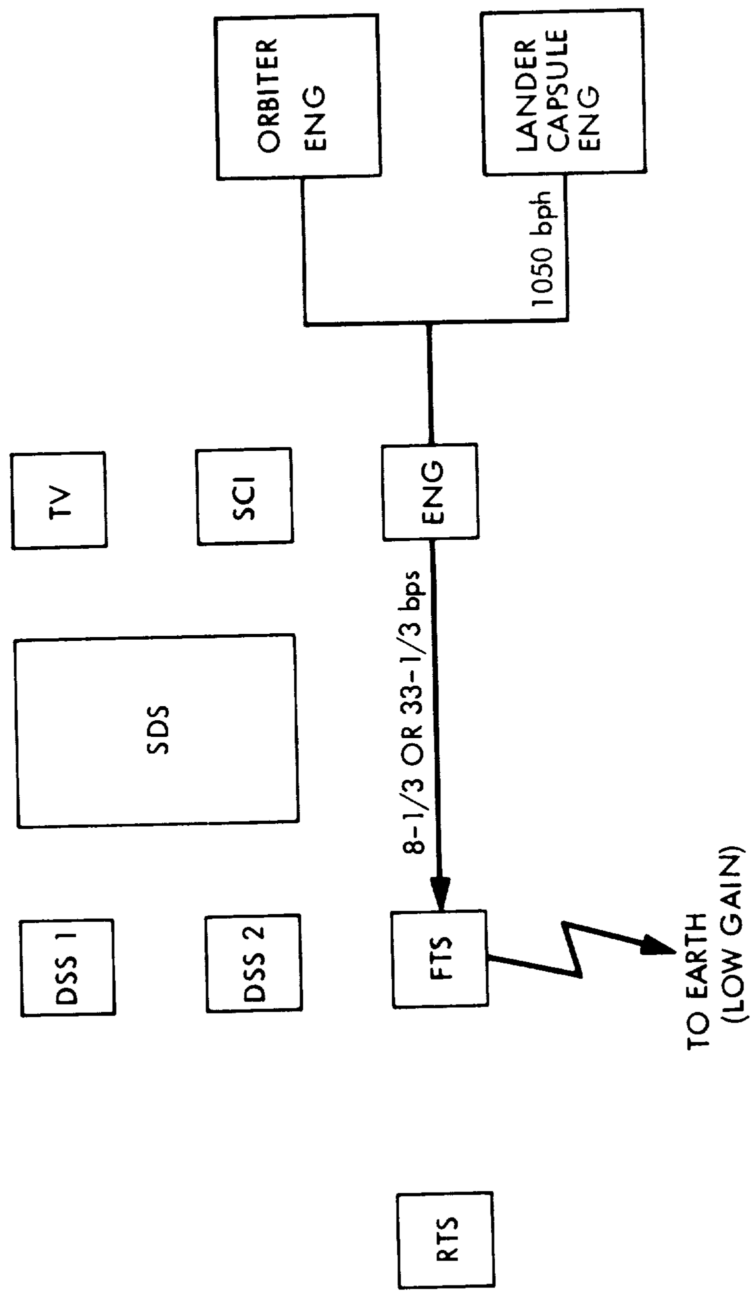

Figure 7F-1. Schematic Representation of Mode 1

1348 bps. See Figure 7F-2.

Mode 2b - Engineering and RF capsule entry relay link being sent in real time during capsule descent while the Earth is in view of orbiter high-gain antenna. This mode requires two channels. Engineering channel is 8 1/3 bps and RF capsule entry relay link is 1348 bps. See Figure 7F-3.

Mode 3a - Engineering being sent in real time over low gain antenna. DSS-2 in record mode. Mode 3a requires a single channel. Hardline capsule checkout is being recorded on DSS-2 since Earth is not in view of high-gain antenna due to pre-separation maneuver. Engineering channel is 8 1/3 bps and hardline capsule checkout is being recorded by DSS-2 at a rate of 1348 bps. See Figure 7F-4.

Since DSS-2 has a capacity of $1.8 \times 10^8 \pm 5\%$ bits, Mode 3a can have a duration of up to,

$$\frac{1.8 \times 10^8 \text{ bits}}{1.348 \times 10^3 \frac{\text{bits}}{\text{second}} \times 3.6 \times 10^3 \frac{\text{second}}{\text{hr}}} \cong 37 \text{ hr}$$

before DSS-2 is filled to capacity (ignoring geometric view constraints).

Mode 3b - Engineering being sent in real time over low-gain antenna. DSS-2 in record mode. Mode 3b requires a single channel. RF capsule checkout relay link is being recorded on DSS-2 since Earth is not in view of high-gain antenna due to orbiter-capsule separation maneuver. Engineering channel is 8 1/3 bps and RF capsule checkout relay link is being recorded by DSS-2 at a rate of 1348 bps. See Figure 7F-5.

Since DSS-2 has a capacity of $1.8 \times 10^8 \pm 5\%$ bits, mode 3b can have a duration of up to 37 hrs before

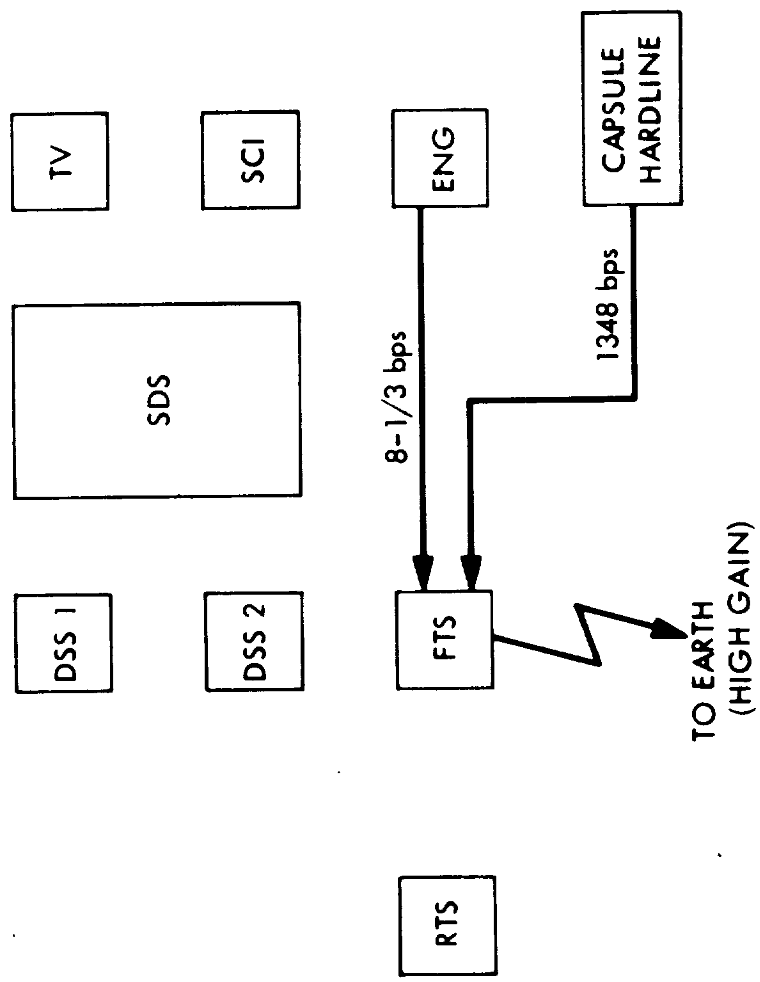

Figure 7F-2. Schematic Representation of Mode 2a

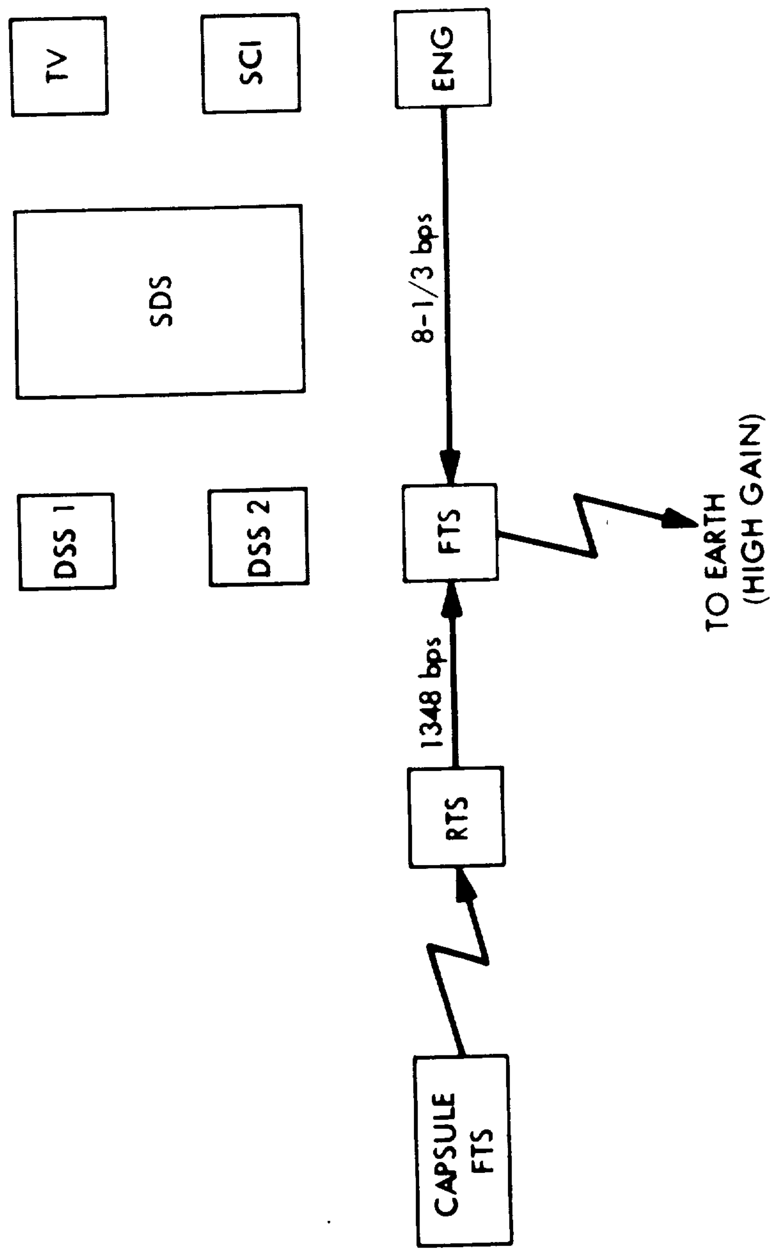

Figure 7F-3. Schematic Representation of Mode 2b

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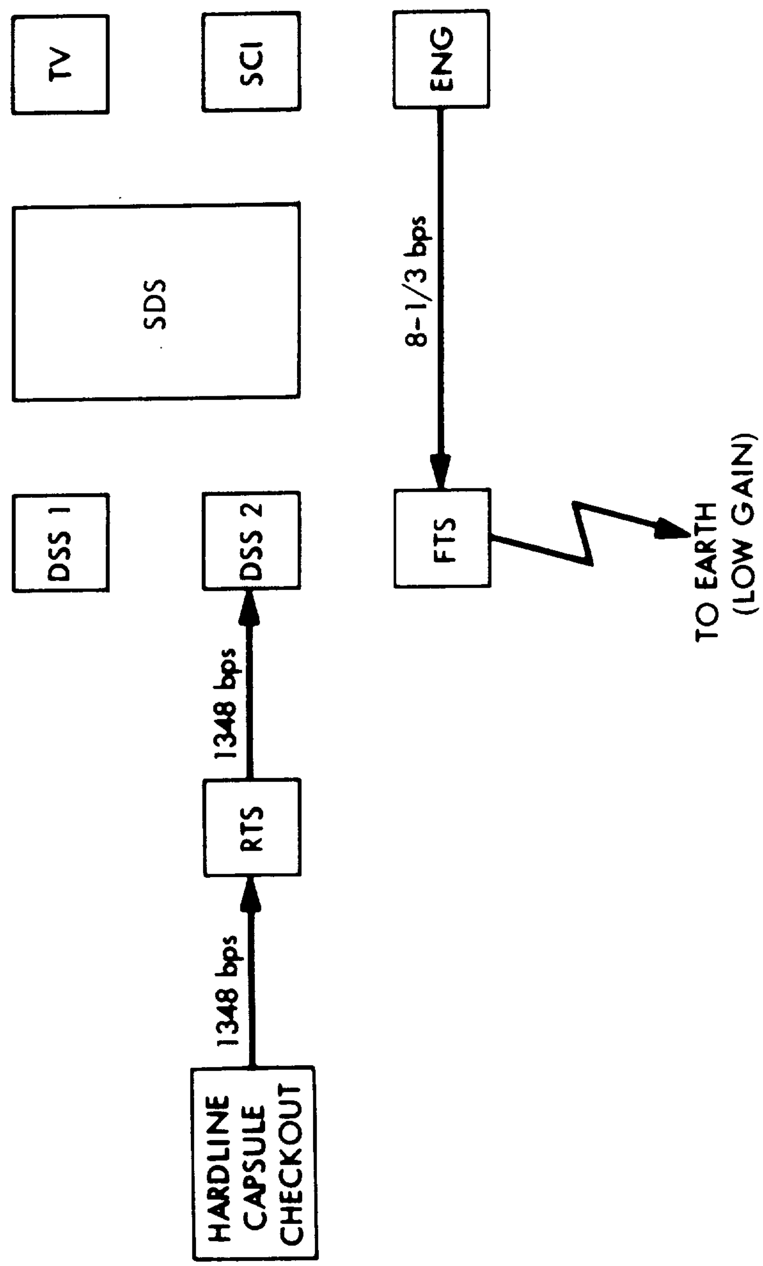


Figure 7F-4. Schematic Representation of Mode 3a

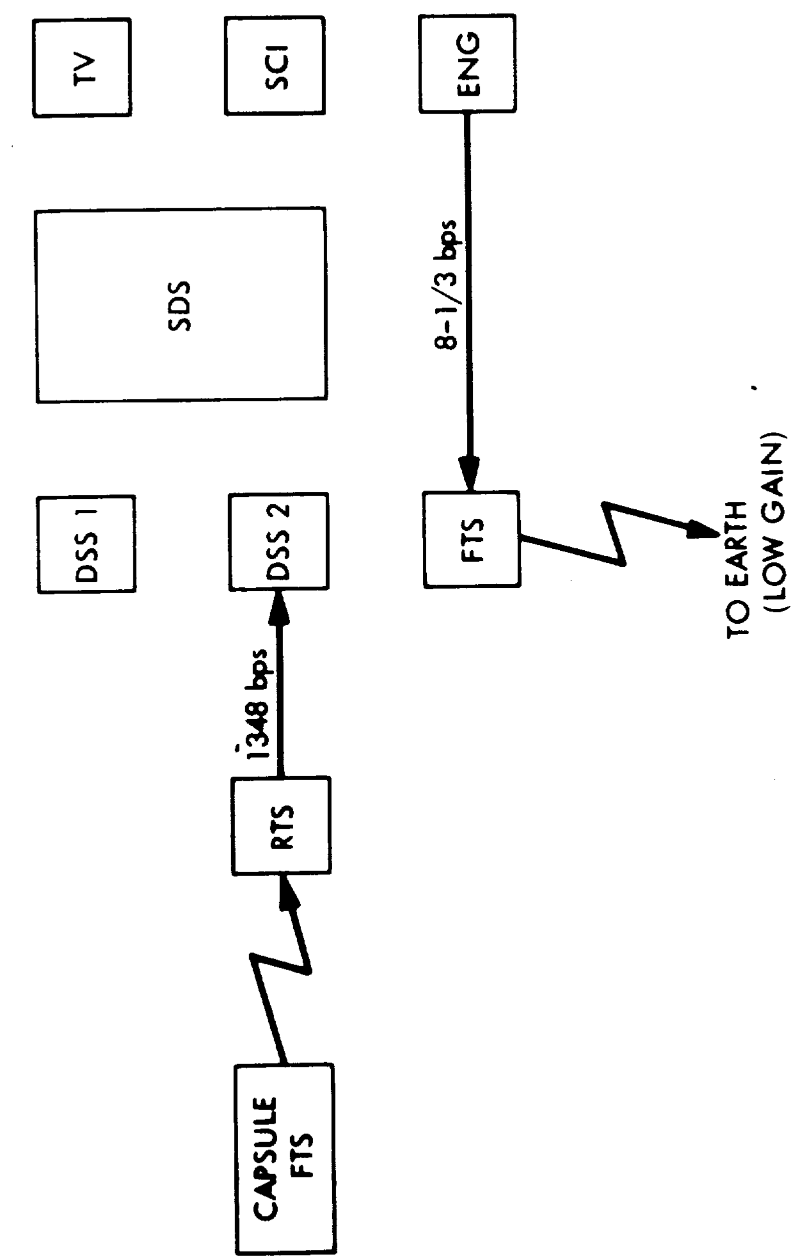


Figure 7F-5. Schematic Representation of Mode 3b

DSS-2 is filled to capacity (ignoring geometric view constraints).

Mode 4 - Engineering being sent in real time over low-gain antenna. DSS-2 in record mode. Mode 4 requires a single channel. Mode 4 will occur during orbiter periapsis passage where RF data from lander is being recorded on DSS-2 at a record rate of 20 Kbps. The engineering data rate is 8 1/3 bps. See Figure 7F-6.

Mode 5 - Engineering is being sent in real time. DSS-1 is in record mode. Mode 5 will require a single channel. Mode 5 will occur at periapsis passage where orbiter TV data and other science is being buffered by SDS and recorded on DSS-1. No orbiter TV or other science is being transmitted in real time. Engineering is being transmitted at 8 1/3 bps and orbiter TV and other science data is being recorded at 132.3 Kbps (MM71 DSS record rate). See derivation of Mode 5 at end of this section. See Figure 7F-7.

Mode 6a - Engineering is being sent in real time. DSS-2 is in playback mode. Mode 6a will require two channels. Mode 6a will occur sometime after periapsis passage where stored capsule data (was recorded previously on DSS-2 at 1348 bps) is played back during the capsule quiet time. The playback will be block-coded high rate at 8.1 Kbps (nominal). The time necessary to unload a single DSS at 8.1 Kbps will be 6.2 hrs. A 210' dish will be required during this time. Engineering will be transmitted at 8 1/3 bps.

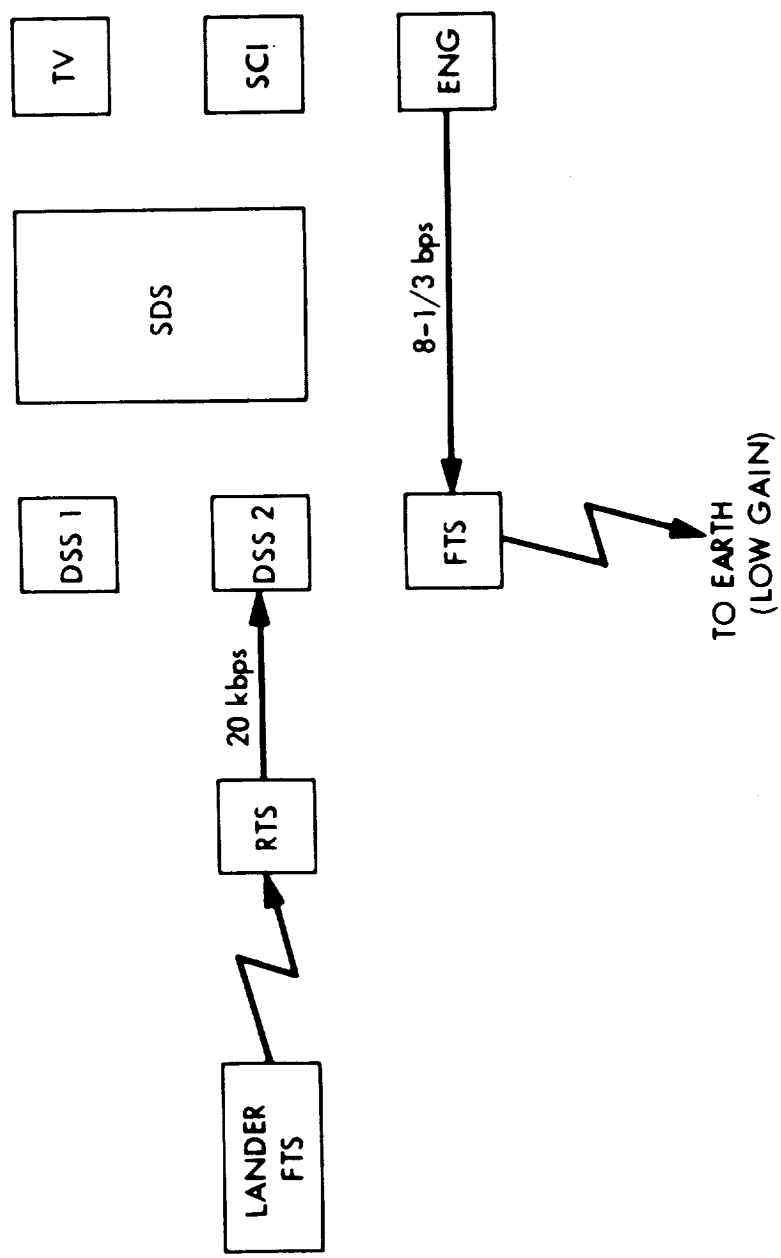


Figure 7F-6. Schematic Representation of Mode 4

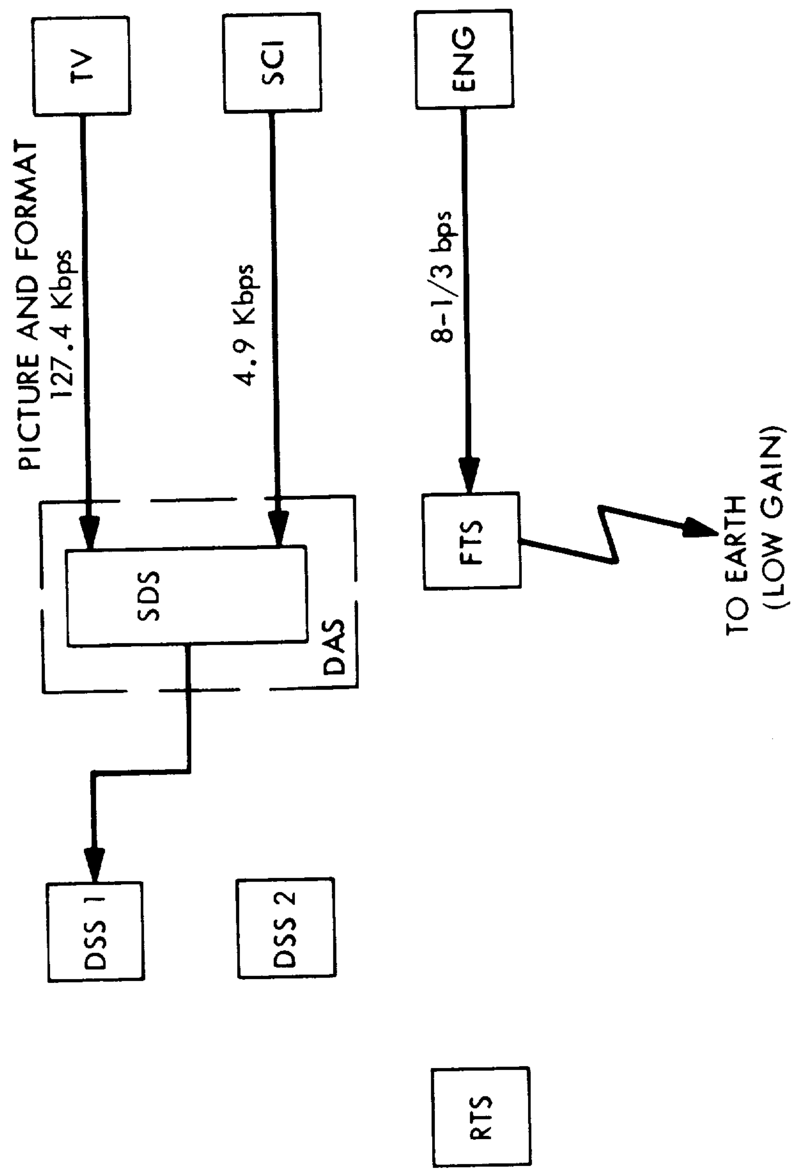


Figure 7F-7. Schematic Representation of Mode 5

Mode 6b - Engineering is being sent in real time. DSS-2 is in playback mode. Mode 6b will require two channels. Mode 6b will occur sometime after periapsis passage where stored lander data (was recorded previously on DSS-2 at 20 Kbps) is played back during the capsule quiet time. The playback will be block-coded high rate at 8.1 Kbps (nominal). The time necessary to unload a single DSS at 8.1 Kbps will be 6.2 hrs. A 210' dish will be required during this time. Engineering will be transmitted at 8 1/3 bps. Figure 7F-8 depicts modes 6a and 6b.

Mode 6c - Engineering is being sent in real time. DSS-1 is in playback mode. Mode 6c will require two channels. Mode 6c will occur sometime after periapsis passage where stored orbiter TV and science data that was obtained in mode 5 is played back during the capsule quiet time. The playback will be block-coded high rate at one of the following (MM71) bit rates - 16.2, 8.1, 4.05, 2.025 or 1.012 Kbps. A 210' dish will be required. Table 7F-1 depicts the maximum time necessary to unload a single DSS at the above mentioned rates. Engineering will be transmitted at 8 1/3 bps. See Figure 7F-9.

Mode 7 - Engineering is being sent in real time at 8 1/3 bps. Low-rate science is being sent in real time at 133 1/3 bps. Mode 7 will require two channels. See Figure 7F-10.

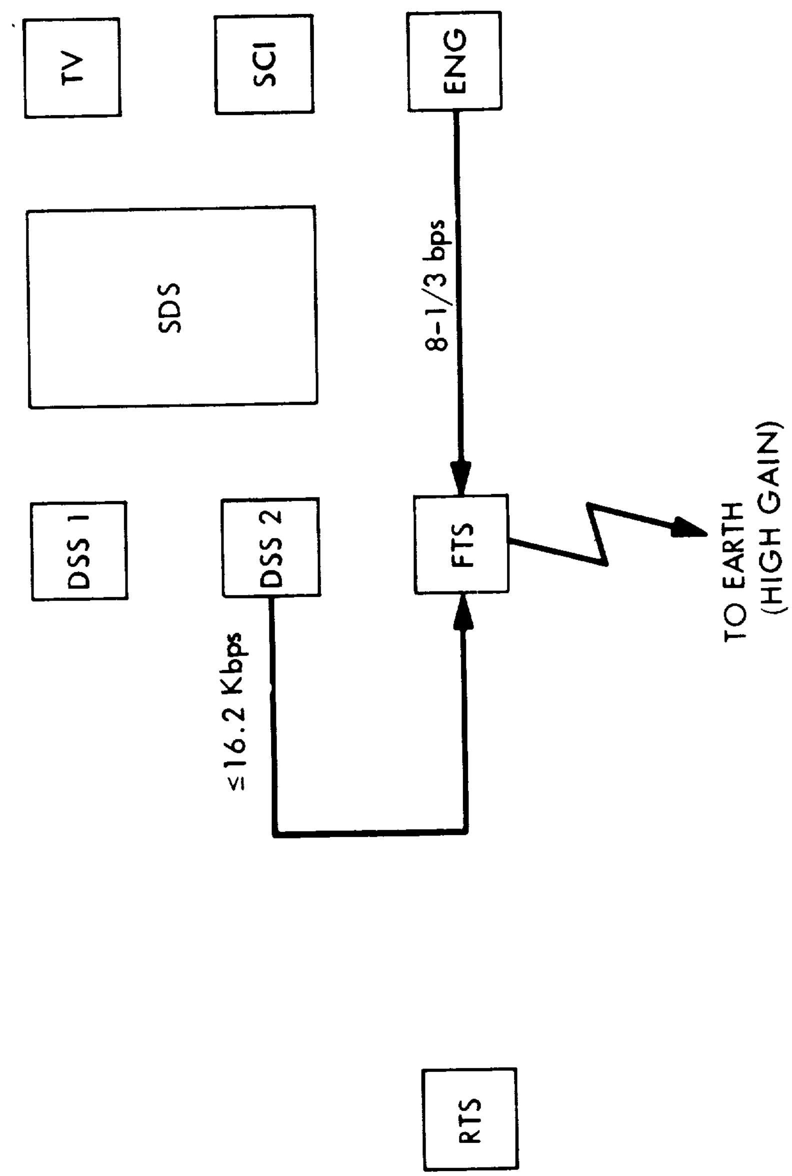


Figure 7F-8. Schematic Representation of Mode 6a & 6b

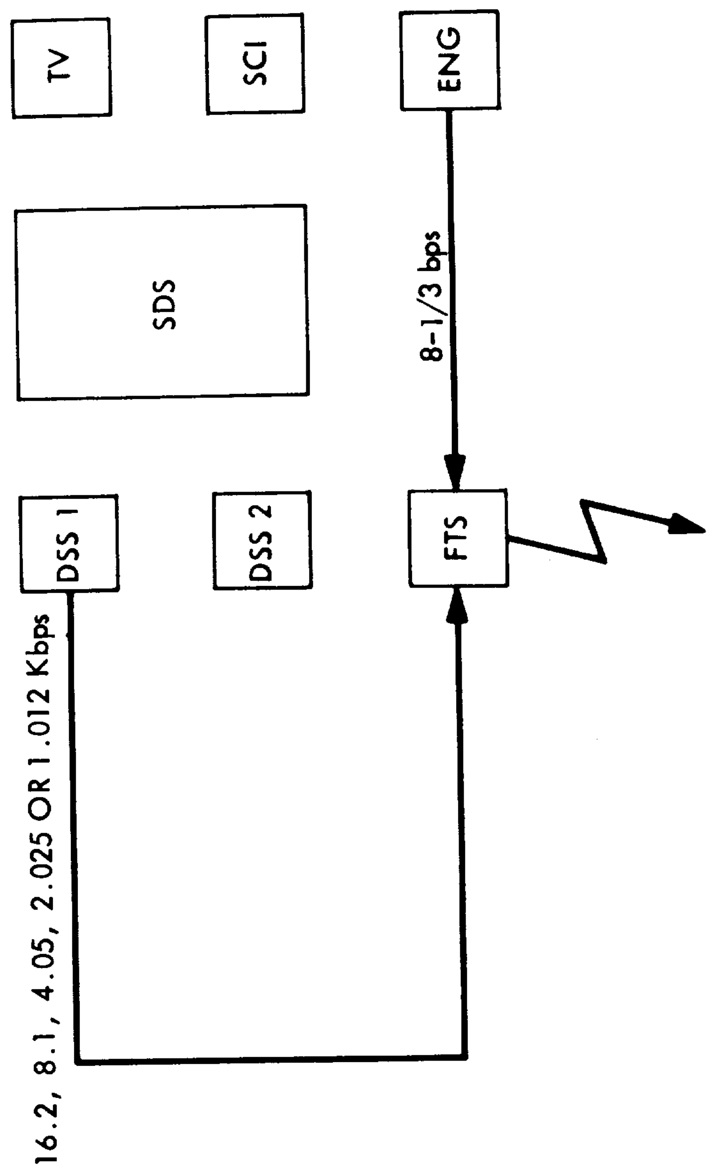


Figure 7F-9. Schematic Representation of Mode 6c

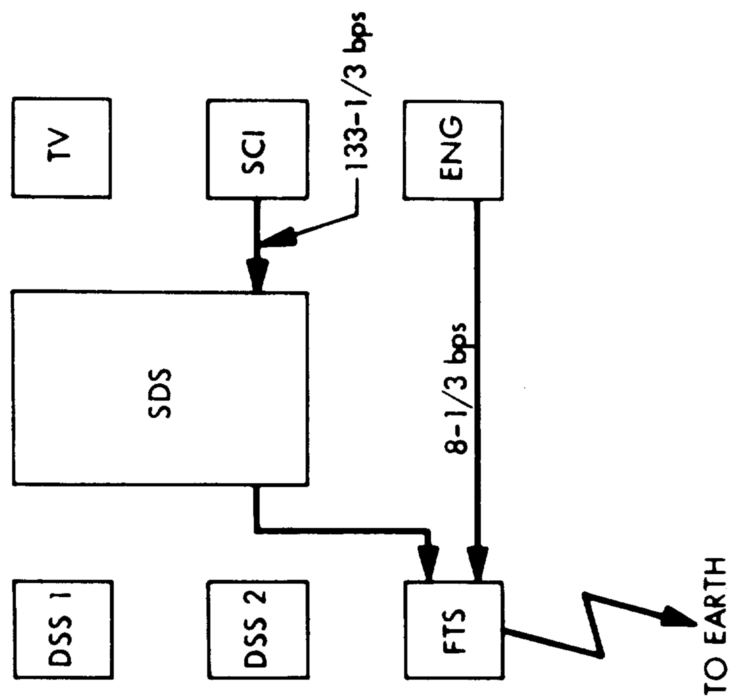


Figure 7F-10. Schematic Representation of Mode 7

Table 7F-1. Playback Times for Orbiter Acquired Data
(Playback time for fully loaded DSS)

Rate	Time
16.2 Kbps	3.1 hrs
8.1 Kbps	6.2 hrs
4.05 Kbps	12.3 hrs
2.025 Kbps	24.7 hrs
1.012 Kbps	49.4 hrs

3. Derivation of Mode 5

In order to discuss this mode some parameters must be initially constrained and the remaining parameters can be derived. One possible method of analysis is as follows:

Step 1 Fix orbiter TV picture size:

$$\left.\begin{array}{l}\text{700 lines}\\ \text{832 pixels/line}\\ \text{9 bits/pixel}\end{array}\right\}\ 5.25 \times 10^6 \text{ bits/picture}$$

This corresponds to the present MM71 TVS.

Step 2 Fix orbiter TV record rate:

This is the rate at which the trace on the vidicon is read into the DSS. For both cameras it is 126 Kbps.

Step 3 Determine orbiter total picture time:

$$\text{Total picture time} = \frac{5.25 \times 10^6 \ \frac{\text{bits}}{\text{picture}}}{126 \times 10^3 \ \frac{\text{bits}}{\text{sec}}} = 41.7 \ \frac{\text{sec}}{\text{picture}}$$

Step 4 Fix maximum periapsis passage science bit rates:

This rate, excluding TVS, is to be buffered by the SDS with TVS data for DSS recording. For the Viking mission the following bit rates are assumed:

Orbiter Science Instruments (Assumed)	
IR Radiometer	1000 bps
IR Spectrometer	2700 bps
IR Mapper	300 bps
TVS	126000 bps
*Contingency	900 bps
Subtotal	130, 900 bps

*Allowance for other instruments, TV engineering, certain status bits, instrument variation, focus, etc.

TV Formatting

Picture count	6 bits/line
Real time frame count	25 bits/line
Line count	10 bits/line
PN sync	31 bits/line
Camera ID	1 bit/line
Contingency	10 bits/line

$$83\ \frac{\text{bits}}{\text{line}} \times \frac{700\ \text{lines}}{\text{picture}} =$$

$$58,100\ \frac{\text{bits}}{\text{picture}}$$

Subtotal $\frac{58,100\ \text{bits/picture}}{41.7\ \text{sec/picture}} = 1400\ \text{bps}$

Total science bit rate required = 132.3 Kbps = Record rate.

Table 7F-2 presents parameters derived from the above four steps. Figures 7F-11 and 7F-12 illustrate the parameters represented in Table 7F-2.

Before implications on an orbiter periapsis sequence can be examined, two other parameters must be fixed, namely:

1) Vidicon erase time = 10 sec (MM71, both cameras)
2) Single DSS capacity = $1.8 \times 10^8 \pm 5\%$ bits (MM71, single DSS)

If one assumes that cameras A & B are alternating and that, while picture information from one camera is being recorded, the vidicon on the other camera is being erased, then the implications are straightforward.

The number of pictures that can be recorded on a single DSS at periapsis passage is:

$$\frac{1.8 \times 10^8\ \text{bits}}{5.52 \times 10^6\ \frac{\text{bits}}{\text{frame}}} = 32.6\ \text{frames} \approx 32\ \text{frames}$$

Table 7F-2. **Data Handling Parameters**

Effective picture record rate	126 Kbps
Total picture time	41.7 sec/picture
Active picture time	39.7 sec/picture
Dead time (flyback) per frame	2 sec
Dead time per line	2.9 ms
Active time per line	56.7 ms
Total line time	59.6 ms
Bits per frame of science other than TV	$.265 \times 10^6$ bits
Bits per frame of TV	5.25×10^6 bits
Total bits per frame	5.52×10^6 bits
Bits per line from vidicon	7488 bits
Bits per line from TV format	83 bits
Bits per line from science instruments	292 bits
Total bits per line	7863 bits

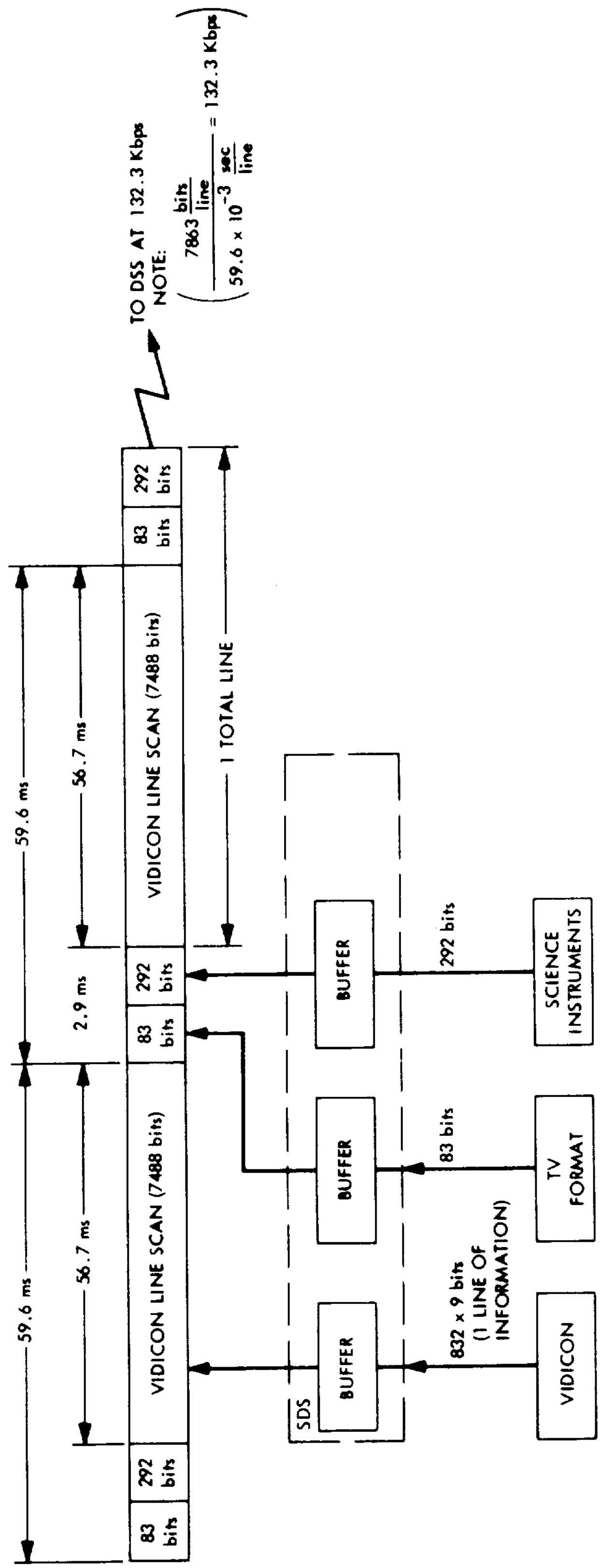


Figure 7F-11. Schematic Representation of Recorded Line

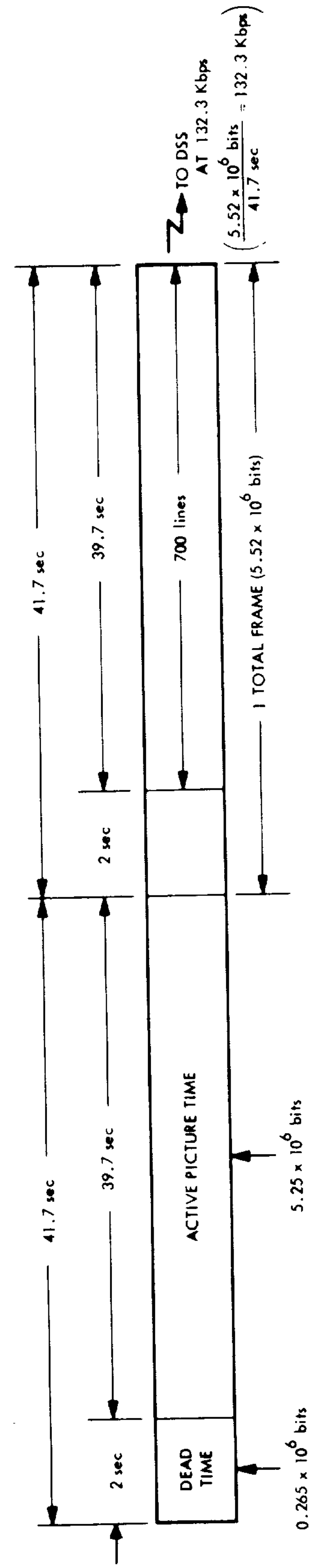


Figure 7F-12. Schematic Representation of Recorded Frame

Since the MM71 DSS has 8 tracks [consisting of four data tracks and four clock (bit sync) tracks], two of which are used for each tape pass, each tape recorder pass can store 8 frames. Figure 7F-13 depicts a single tape recorder pass for alternating cameras.

For the case of one camera (either A or B) taking a succession of pictures, the tape recorder will be shut off during the 10-second vidicon erase time. One second of tape will be lost by turning on and off the tape recorder during the 10-second erase time. Therefore, if we assume that 7 seconds of tape are lost for 1 track, we can derive the number of bits lost per track and the DSS storage capacity in the following manner:

Assume 8 frame/pass $\Rightarrow$ 7 sec of tape lost if cameras cycle continuously.

$$\text{Bits lost/pass} = 7 \text{ sec/pass} \times 132.3 \times 10^3 \frac{\text{bits}}{\text{sec}} = 926 \times 10^3 \frac{\text{bits}}{\text{pass}}$$

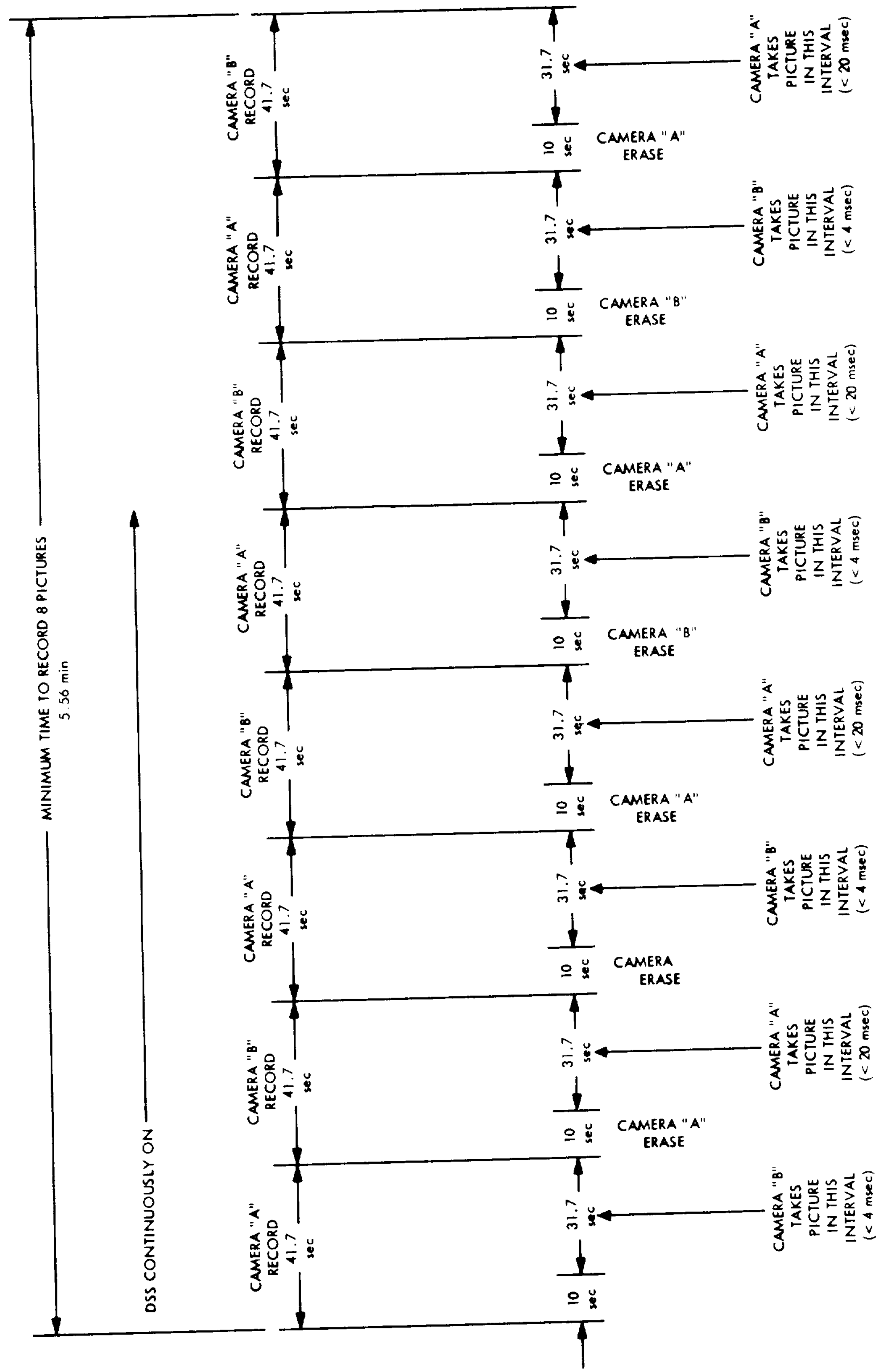
$$\text{Total bits/pass} = \frac{1.8 \times 10^8 \text{ bits}}{4 \text{ pass}} = 4.5 \times 10^7 \text{ bits (nominal)}$$

$$\text{Effective bits/pass} = (4.5 - .0926)\, 10^7 \frac{\text{bits}}{\text{pass}} \cong 4.4 \times 10^7 \frac{\text{bits}}{\text{pass}}$$

$$\text{Frames/pass} = \frac{4.4 \times 10^7 \frac{\text{bits}}{\text{pass}}}{5.52 \times 10^6 \frac{\text{bits}}{\text{frame}}} \cong 8.0 \frac{\text{frames}}{\text{pass}}$$

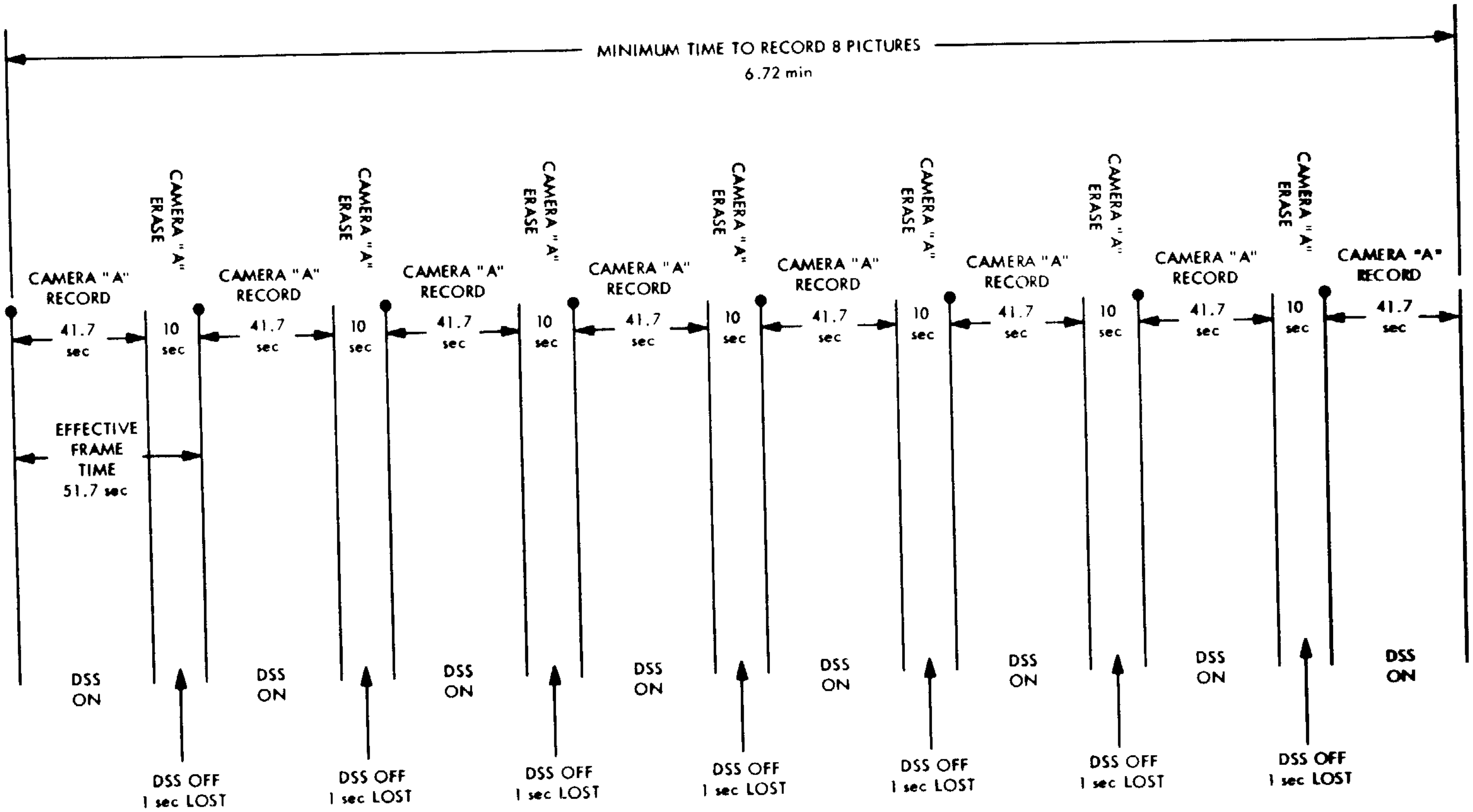
Thus the assumption of 8 frames/pass is justified if we assume the nominal case. Figure 7F-14 depicts the above case where one camera is taking a succession of pictures.

A typical sequence of orbiter pictures at periapsis is most likely to be a combination of Figures 7F-13 and 7F-14, i.e., a succession of B frames interspersed with A frames. However, the bounds on filling a single DSS to capacity lies between 22.3 minutes and 27 minutes.



NOTE: MINIMUM TIME TO RECORD 32 PICTURES (CAPACITY OF ONE DSS) ≅ 22.3 min

Figure 7F-13. Frame Sequence for One Track -- Alternating Cameras



NOTE:

1. 7 seconds OF TAPE LOST
2. DOTS INDICATE WHEN PICTURE IS BEING TAKEN (TAKES ≤ 20 msec) THIS HAS NOT BEEN TAKEN INTO ACCOUNT BUT IS ASSUMED PART OF ERASE TIME OF 10 sec
3. ORBIT SCIENCE IS LOST DURING 10 second GAP UNLESS SENT IN REAL TIME (ANOTHER MODE). HOWEVER, SCIENCE MAY ONLY BE ON WHEN TV IS BEING RECORDED OR LOSS MAY BE CONSIDERED NEGLIGIBLE
4. MINIMUM TIME TO RECORD 32 PICTURES (CAPACITY OF 1 DSS) ≅ 27 min

Figure 7F-14. Frame Sequence for One Track -- Same Camera (A or B)

G. TELECOMMUNICATIONS SYSTEM DESCRIPTION

1. Functional Description

The Viking Orbiter telecommunication system essentially provides for data transfer along the relay link between the landing capsule and the orbiter and along the direct link between the orbiter and the Deep Space Network (DSN) tracking station. A block diagram of the different links involved is shown in Figure 7G-1. Before capsule separation a multiconnector cable connects the capsule and the orbiter. When this line is cut, communication continues through a 400-MHz RF link.

A 400-MHz beacon on the orbiter will transmit an AM turn-on command every 60 seconds or so. When the landing capsule is within range and receives a strong command, a playback of accumulated lander telemetry data will begin. Additional turn-on commands will not be sent by the orbiter until this data playback is completed. Playback is by means of variable data rate FSK (frequency shift keying).

A main telemetry link which carries data from the orbiter to the ground operates at 2297 MHz. This link operates independently of the 400-MHz equipment. It can also operate independently of the 2116-MHz command link. When two-way doppler tracking is required, both of these S-band channels are used simultaneously. A two way ranging channel is also available by using both the 2297-MHz and 2116-MHz frequencies simultaneously.

2. Interfaces

a. Launch Vehicle/Spacecraft Interface

This interface is of a very temporary nature and disappears after separation. When the launch vehicle is on the pad, data from the spacecraft can be monitored directly through the umbilical connectors. Immediately after launch a cable will still transfer spacecraft data to the launch vehicle. Problems associated with this interface are minimized because most of the time it is accessible on the ground. Even at injection, after the direct connection is severed, communication is simplified because signal

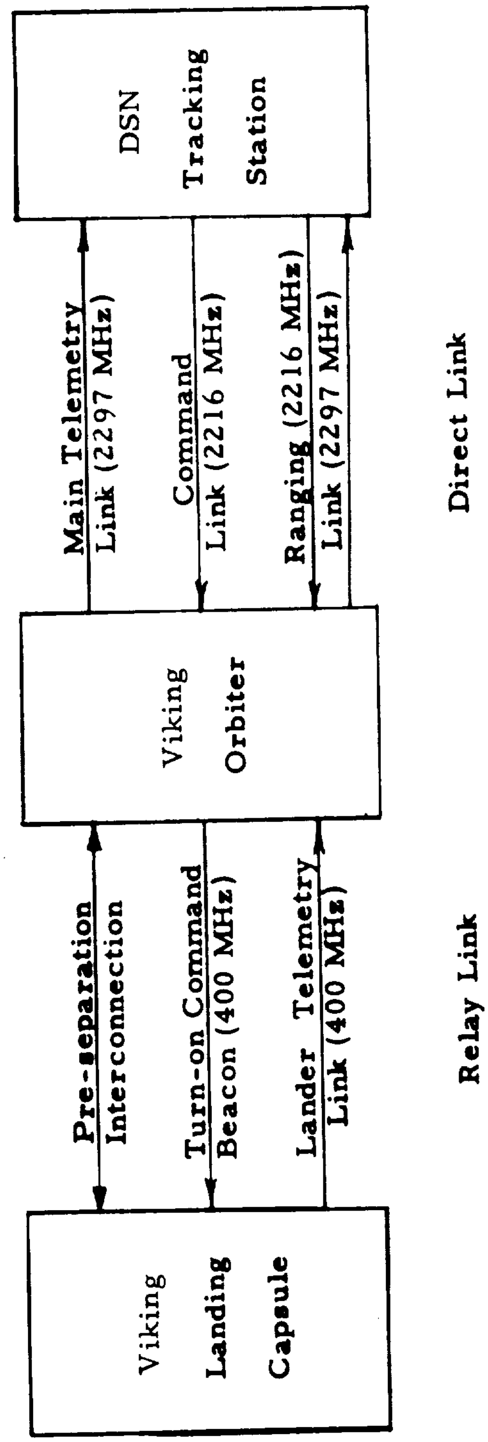


Figure 7G-1. Block Diagram of Viking Telecommunication System levels are strong and there is relatively little data.

b. Spacecraft/DSN Interface

Since the Viking Mars Orbiter design and the DSN tracking stations are both managed by the Jet Propulsion Laboratory, special care can be taken to insure the compatibility of all parts of the direct link system. Radio frequencies, modulation, and coding will be standardized. Also, the antenna gains, transmitter powers, and other link parameters will be carefully matched to maintain sufficient margin for the required performance.

Perhaps the most difficult part of this interface is in scheduling the DSN tracking stations. Figure 7G-2 illustrates DSN loading by plotting for each mission the time of day for local meridian crossing against the day for calendar years 1973 and 1974. An Explorer mission which runs almost concurrently will mainly use the 85-foot antennas with their separate receivers. More severe conflict may occur near noon with several other missions. Ample tracking time should be available later on in the evening however.

Table 7G-1 compares the parameters of the 85-foot and 210-foot DSN tracking stations. It is likely that the larger antennas will be used during most of the mission. A similar table could also be prepared for the spacecraft but due to the large number of adjustable parameters it is easier to list them in the appropriate Design Control Tables of the next section. In all cases possible parameters from the original functional specifications have been replaced by measured test results.

c. Relay Link Interface

By far the most difficult interface problem is in the relay link. The major hardware blocks making up the Viking relay system are the antennas, the lander and orbiter (beacon) transmitters, the orbiter and lander (beacon) receivers, and the orbiter relay data demodulation, data handling and mode control circuits. Also a part of any relay link consideration is the "channel" which includes such factors as antenna pointing,

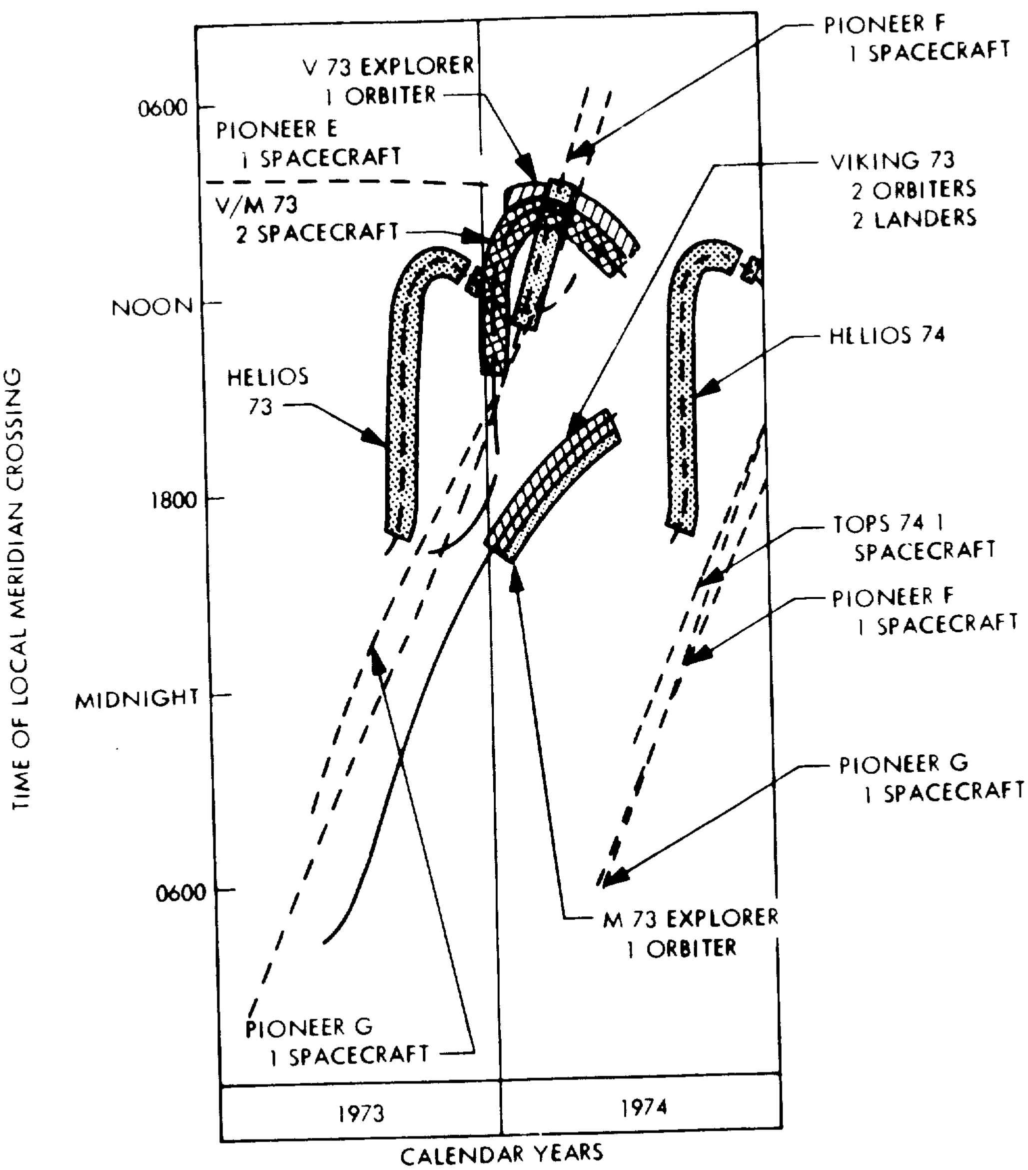


Figure 7G-2. DSN Loading

Table 7G-1. DSN Tracking Station Parameters and Tolerances

Parameter and Units	85-Foot Antenna			210-Foot Antenna		
	Nominal	Favorable	Adverse	Nominal	Favorable	Adverse
Transmitting antenna gain, db	51.8	0.9	-0.9	59.5	0.6	-0.6
Transmitting antenna ellipticity, db	1.0	-0.3	0.3	2.4	-0.4	0.4
Transmitting power, dbm	70.0	0.5	0.0	86.0	0.5	-0.0
Receiving antenna gain, db	53.3	0.6	-0.6	61.4	0.4	-0.4
Receiving antenna ellipticity, db	0.4	-0.1	0.1	0.2	-0.1	0.1
Receiver Noise Temperature in deg. K (near 25 degrees elev.) Listen Transmit	50.0	-6.0	6.0	35.0	-1.0	3.0
Carrier Threshold Noise Bandwidth Hz, db	10.8	-1.0	0.0	10.8	-1.0	0.0

communication distance, multipath effects and so on. A more detailed block diagram of the relay system is presented in Figure 7G-3.

3. Orbiter Telecommunications System

The Viking telecommunication system will essentially consist of a Mariner 1971 S-band telecommunication system with a minimum number of modifications. A new 400-MHz relay system is added to allow for communication with the lander. Figure 7G-4 gives a block diagram showing the details of the system.

One of the ways to improve the performance of the Mariner 1971 radio system is to use larger and more antennas. Table 7G-2 lists the five different antennas which will be carried on the orbiter. It also indicates the general type of antenna, its position, and when it is used.

Each of the antennas adds a small amount of weight to the total. These weights are included in the relay radio and radio frequency weights shown in Table 7G-3. A final column in this table shows the power necessary to operate this equipment. Since the relay link turn-on beacon operates only momentarily, the 13 watts is not a continuous load. Also, only one tape recorder is operated at a time reducing the data storage power consumption to 22 watts.

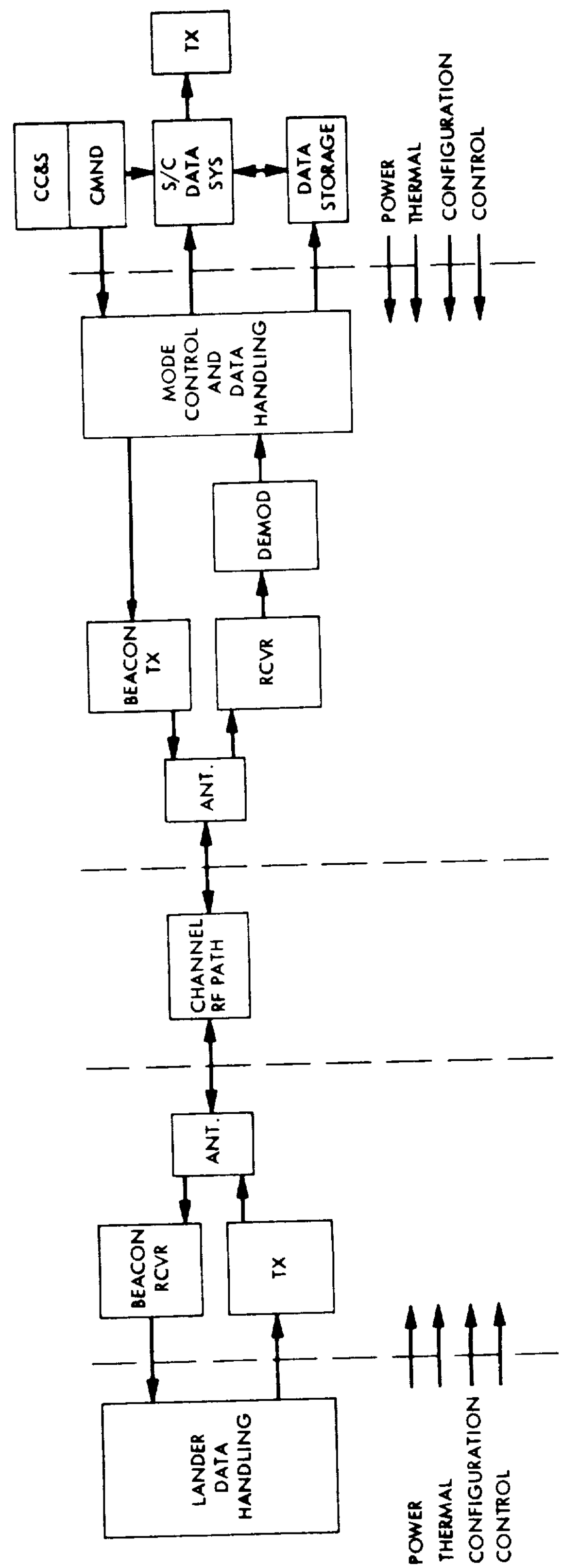


Figure 7G-3. Relay Link Block Diagram

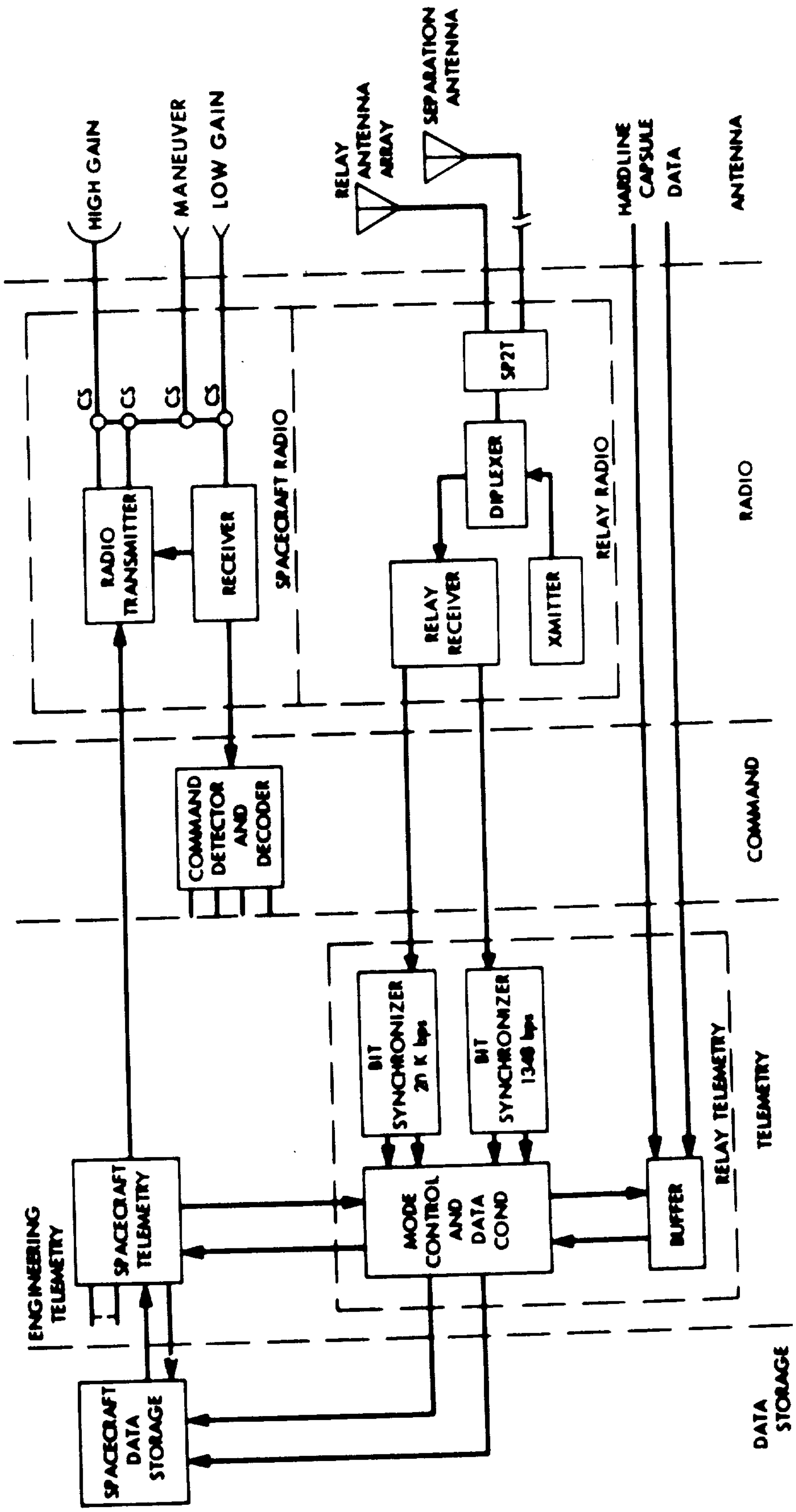


Figure 7G-4. Viking Orbiter Telecommunications

ANTENNA	TYPE	POINTING DIRECTION
Low Gain Antenna	Horn (Same as M'71)	0.0 Degrees Cone
Maneuver Antenna	Horn (Same as M'71)	To be determined (Somewhat to one side of 180.0 degrees cone.)
High Gain Antenna	Steerable 5 ft diameter Parabolic Reflector	Adjustable
Relay Antenna	Array	138 degrees cone 259 degrees clock
Separation Antenna	Probe in Bioshield Base	180.0 degrees cone

FOLDOUT FRAME /

RANGE OF MOTION	RECEIVES	TRANSMITS
Fixed	Commands throughout mission. Uplink for two-way doppler and ranging throughout mission.	Telemetry before encounter. Downlink for two-way doppler and ranging before encounter,
Fixed	During maneuvers when spacecraft has lander capsule side toward sun.	During maneuvers when spacecraft has capsule side toward sun.
Initial Final 38.8 17.9 degrees cone 274.1 256.8 degrees clock		Downlink for two-way doppler and ranging after encounter.
Fixed	Telemetry from VLC and lander.	Turn-on command
Fixed (Jettisoned with Bioshield Base)	Telemetry from capsule before Bioshield Base is jettisoned.	Never

Table 7G-2. Viking Orbiter Antennas

Table 7G-3. Viking Orbiter Telecommunications Physical Parameters Summary

SUB-SYSTEM	WEIGHT (lbs)	POWER (watts)
RELAY RADIO with antenna	20.4	13
RADIO FREQUENCY with antennas	74.4	123
COMMAND	11.7	3.5
RELAY TELEMETRY	10.	5.0
FLIGHT TELEMETRY	26	15.0
DATA STORAGE	40.0	22
TOTAL	182.5	181.5

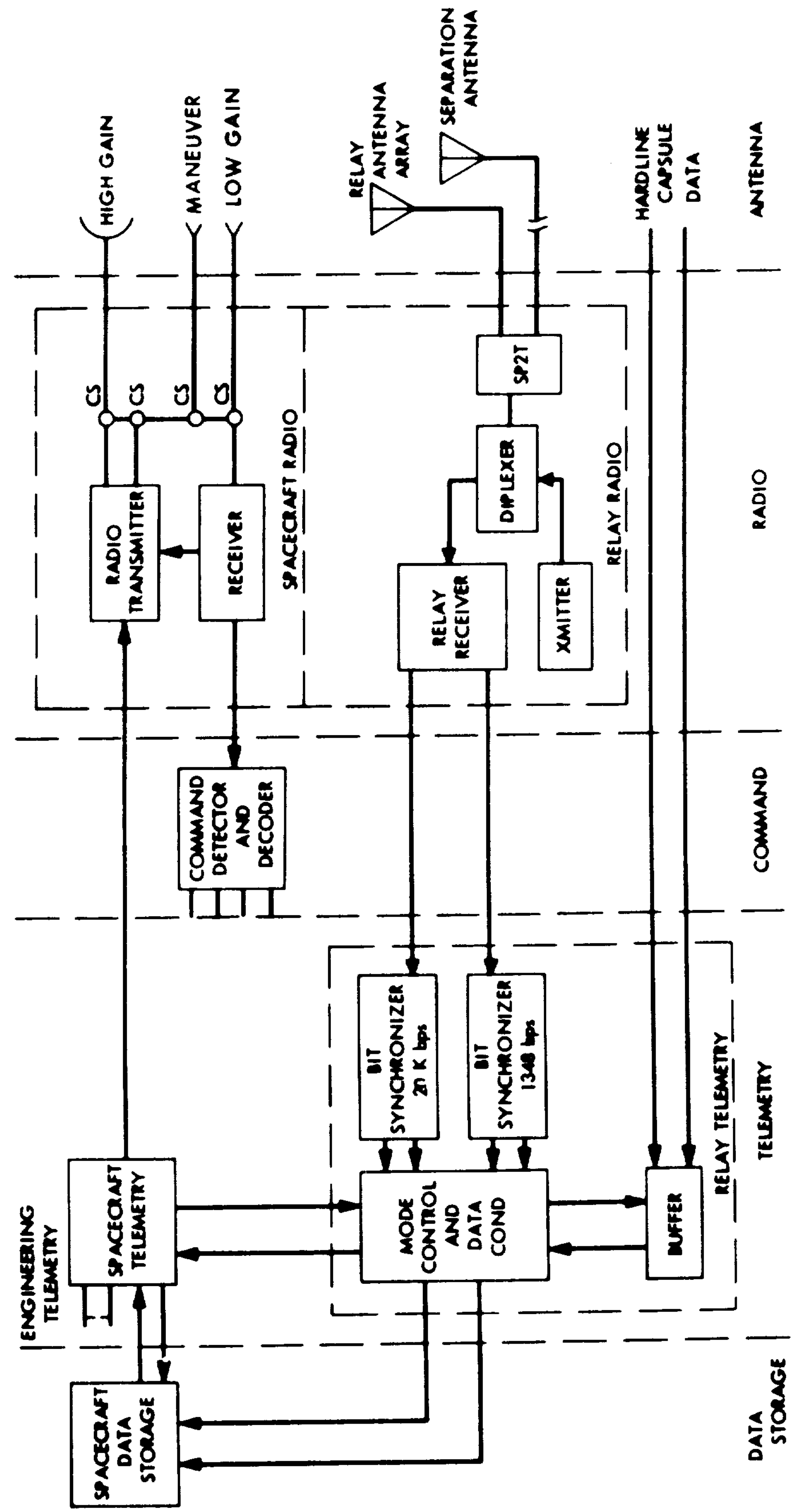


Figure 7G-4. Viking Orbiter Telecommunications

H. ORBITER TELECOMMUNICATION REQUIREMENTS AND PERFORMANCE PREDICTIONS

1. General Requirements

There are certain restrictions placed upon the orbiter telecommunication system by the basic data requirements and the available bit rates. These, together with the mission geometry, usually dictate the antenna and radio configurations. Table 7H-1 has been included to summarize telemetry modes during each phase of the mission. A total of seven different modes of operation have been distinguished and several have alternate bit rates available.

2. Performance Predictions

a. Direct Link Telemetry

The telemetry link carries engineering telemetry, doppler, ranging, and relay data from the orbiter to Earth. An analysis of this link has been made for both trajectory A and trajectory B. Results are given in Figures 7H-1 and 7H-2, respectively. A data rate of 33 1/3 bps can be maintained to well over 150 days from launch with the low gain spacecraft antenna and a 210-foot DSN antenna. After this the bit rate must be reduced to 8 1/3 bps or the high gain spacecraft antenna must be used.

A curve showing telemetry link performance after the spacecraft begins to orbit Mars is shown in Figure 7H-3. A useful period of 100 days has been labeled beginning at encounter for both trajectory A and trajectory B. During this time block coded data will be played back at a high rate. It should be possible to maintain a 4 Kbps rate throughout this phase of the mission.

During orbit insertion and trim maneuvers, engineering telemetry is transmitted over the maneuver antenna having a pattern directed toward the +Z direction of the spacecraft. This antenna is placed at a 180-degree cone angle. Figure 7H-4 shows the performance expected from this antenna when it is pointed directly at the Earth.

Table 7H-1. Viking Orbiter Telecommunications Links, Data Requirements and Bit Rates

TABLE 7H-1

MODE	TIME PERIOD OR MISSION PHASE	DIRECT LINK: DATA REQUIREMENT	DIRECT LINK: TRANSMITTER POWER, w	DIRECT LINK: DATA RATE bps	DIRECT LINK: RECEIVER	RELAY LINK: DATA REQUIREMENT	RELAY LINK: TRANSMITTER	RELAY LINK: DATA RATE, bps	RELAY LINK: RECEIVER
1	CRUISE BEFORE ENCOUNTER	ENGINEERING DATA ONLY	20	33-1/3 OR 8-1/3	$35^\circ\ ^{+3^\circ}_{-1^\circ}$	SUB-COMMUTATED	HARD LINE	1050[1]	HARD LINE
1	MANEUVERS	ENGINEERING DATA ONLY	20	8-1/3	$35^\circ\ ^{+3^\circ}_{-1^\circ}$				
2a	CAPSULE CHECKOUT	ENGINEERING DATA AND FEEDTHROUGH	20	8-1/3 AND 1348	$35^\circ\ ^{+3^\circ}_{-1^\circ}$	FEED THROUGH	HARD LINE	1348	HARD LINE
2b	LOW RATE RELAY	ENGINEERING DATA AND FEEDTHROUGH	20	8-1/3 AND 1348	$35^\circ\ ^{+3^\circ}_{-1^\circ}$	COAST DATA	5 w	1348	NON COHERENT FSK 3 db NF
2b	CAPSULE ENTRY (ON SUN)	ENGINEERING DATA AND FEEDTHROUGH	20	8-1/3 AND 1348	$35^\circ\ ^{+3^\circ}_{-1^\circ}$	ENTRY DATA	5 w	1348	NON COHERENT FSK 3 db NF
3a,b	CAPSULE ENGINEERING (OFF SUN)	ENGINEERING DATA ONLY	20	8-1/3	$35^\circ\ ^{+3^\circ}_{-1^\circ}$	ENTRY DATA	HARD LINE OR 5 w	1348	RECORDED
4	LANDED DATA	ENGINEERING DATA ONLY	20	8-1/3	$35^\circ\ ^{+3^\circ}_{-1^\circ}$	10^7 bits/pass	50 w	20K	RECORDED
5	ORBIT SCIENCE	ENGINEERING DATA WHILE RECORDING SCIENCE	20	8-1/3	$35^\circ\ ^{+3^\circ}_{-1^\circ}$				
6 a, b, c	PLAYBACK STORED DATA	ENGINEERING DATA AND RECORDED DATA	20	8-1/3 bps AND 16, 8, 4, 2, 1 kbps	$35^\circ\ ^{+3^\circ}_{-1^\circ}$				
7	LOW RATE SCIENCE	ENGINEERING DATA AND LOW-RATE SCIENCE	20	8-1/3 bps AND 133-1/3 bps	$35^\circ\ ^{+3^\circ}_{-1^\circ}$				

[1] Bits per hour

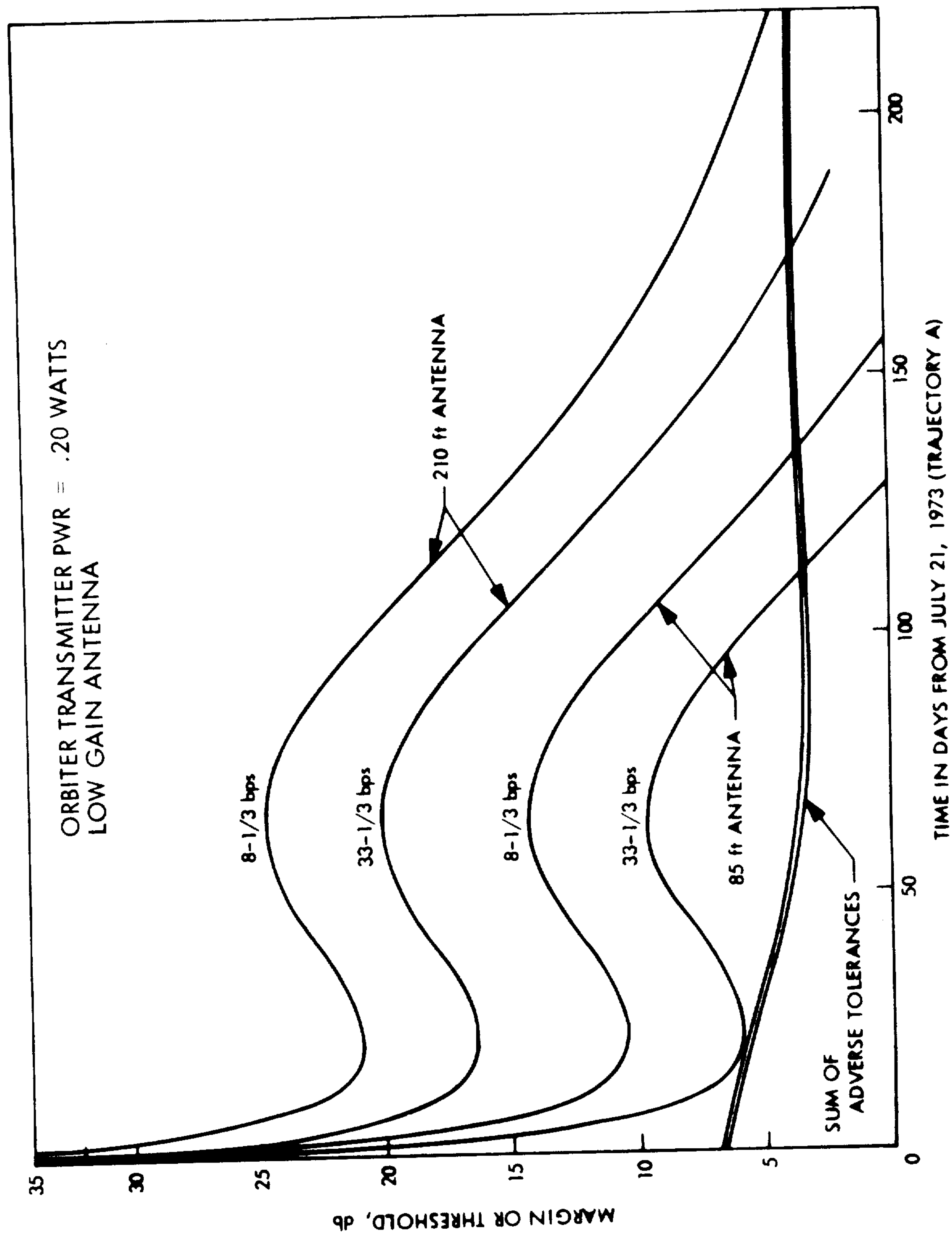


Figure 7H-1. Telemetry Performance During Cruise -- Trajectory A

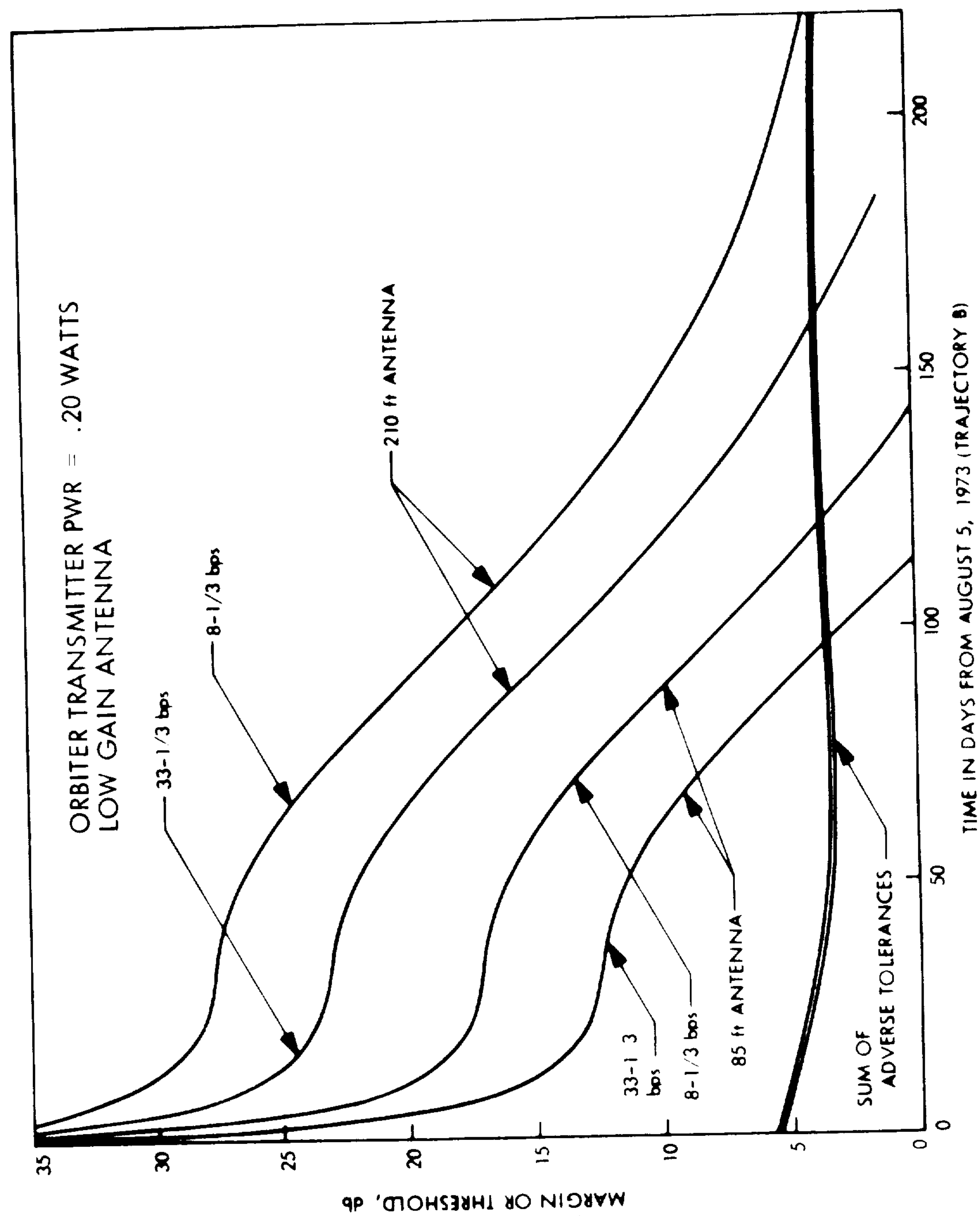


Figure 7H-2. Telemetry Performance During Cruise -- Trajectory B

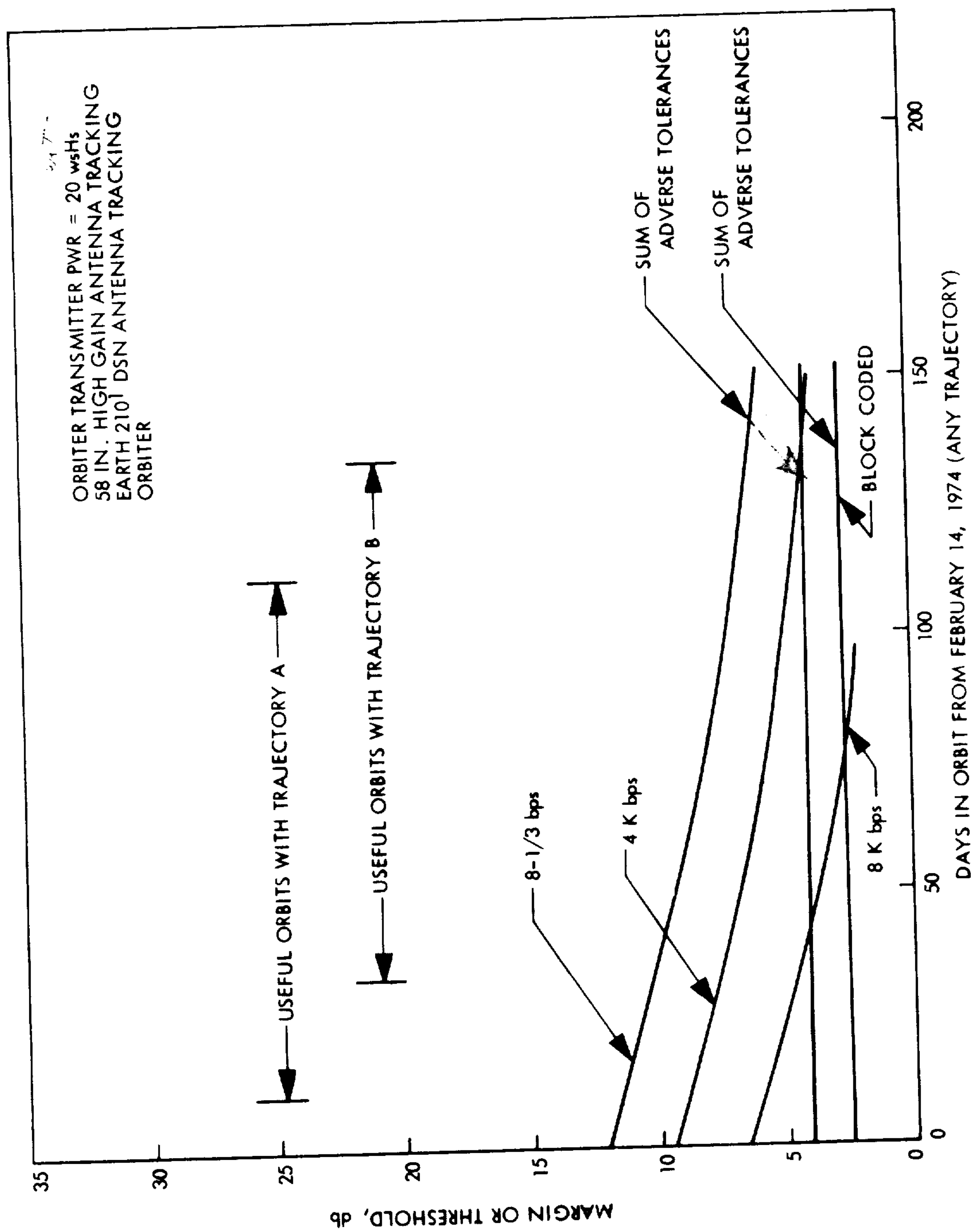


Figure 7H-3. Telemetry Performance During Orbit -- 58" Antenna

FIG. 7H-4

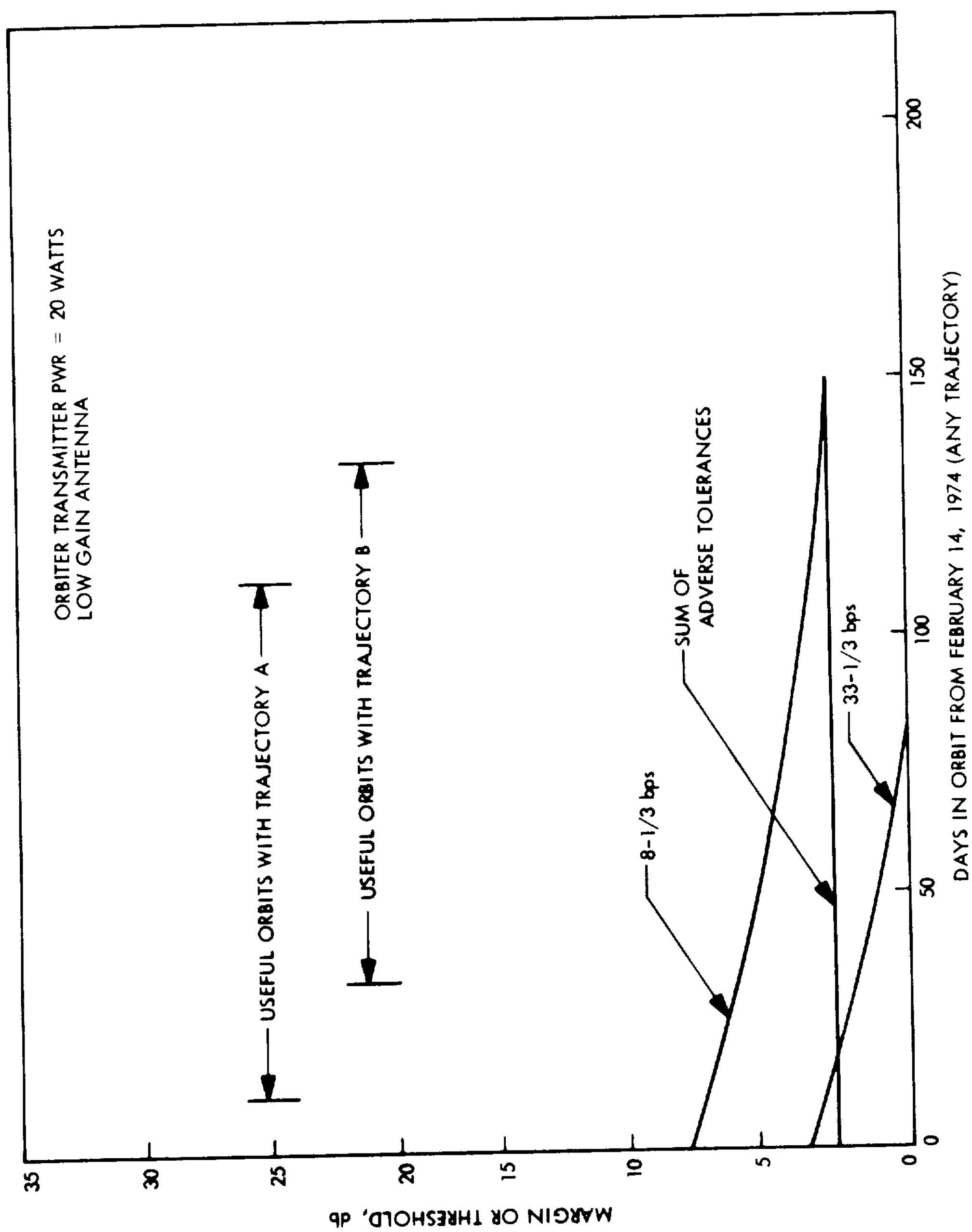


Figure 7H-4. Telemetry Performance During Orbit -- Maneuver Antenna

Since a pointing error of 30 degrees reduces antenna gain by about 1 db, it may be important to have the maneuver antenna pointed accurately at the Earth. For this reason the final maneuvers of the spacecraft may require an extra step. First, rotations about the roll and pitch axes will orient the engine in the proper direction for its burn. Then a rotation about the roll axis will be made to point the antenna as close to the Earth as possible. This should extend the engineering telemetry coverage with the maneuver antenna to most of the spacecraft +Z direction hemisphere.

Telemetry Design Control Tables 7H-2 and 7H-3 have been included to illustrate typical parameters for low rate and high rate telemetry. These values and tables such as these were used to generate the preceding curves.

b. Direct Link Command

This command link is used for two-way doppler and ranging as well as for commands. Because of the low data rate (1 bps), the command performance remains well above threshold throughout the early cruise phase of the mission. Figure 7H-5 shows the performance of the command link after the spacecraft begins to orbit Mars. For convenience, 100-day intervals have been indicated starting from the arrival dates of trajectory A and trajectory B.

With a 400-KW transmitter and a 210-foot antenna, a large command margin is available throughout the mission. It will also be possible for the spacecraft to receive commands on the maneuver antenna. For this case the performance margin should be as good or better than that of Figure 7H-5. Thus, commands can be transmitted to a spacecraft in any orientation throughout the entire mission. A typical Design Control Table, 7H-4, is shown to illustrate the command link parameters.

c. Direct Link Ranging

Ranging is an important mode of operation for which performance calculations have not yet been made. Work on ranging performance curves and design control tables is now in progress.

TABLE 7H-2. CRUISE MODE I ENGINEERING TELEMETRY

TELEMETRY DESIGN CONTROL TABLE

TELEMETRY MODE: CRUISE TELEMETRY DATE: 1-31-69

SCIENCE CHANNEL: ______ ENGINEERING CHANNEL: 8-1/3 OR 33-1/3 BPS STATION: ______

210 FOOT TRACKING STATION; LOW GAIN ANTENNA AT 0.0 CONE

LAUNCH: 2-21-73 ARRIVAL: 2-25-74

TIME IN MISSION: DATE 2-25-74 HOURS FROM INJECTION: 5256

NO.	PARAMETER	NOMINAL VALUE	TOLERANCE		
			FAVORABLE	ADVERSE	
1	TOTAL TRANSMITTER POWER (DBM)	42.32	+0.50	-0.50	
2	TRANSMITTING CIRCUIT LOSS (DB)	-1.45	+0.25	-0.25	
3	TRANSMITTING ANTENNA GAIN (DB)	7.25	+0.40	0.40	
4	TRANSMITTING ANTENNA POINTING LOSS (DB)	-2.51	+0.42	-0.42	
5	SPACE LOSS (DB) F = 2297 MHz, R = 205.53 x 10^6 KM	-265.93	0.00	0.00	
6	POLARIZATION LOSS (DB)	-0.11	+0.03	-0.03	
7	RECEIVING ANTENNA GAIN (DB)	61.40	+0.40	-0.40	
8	RECEIVING ANTENNA POINTING LOSS (DB)	0.00	0.00	0.00	
9	RECEIVING CIRCUIT LOSS (DB)	0.00	0.00	0.00	
10	NET CIRCUIT LOSS (DB) (2+3+4+5+6+7+8+9)	-201.35	+1.50	-1.50	
11	TOTAL RECEIVED POWER P(T) (DBM) (1+10)	-159.03	+2.00	-2.00	
12	RECEIVER NOISE SPECTRAL DENSITY (DBM/HZ) ELEV 25 DEG	-183.16	-0.13	+0.36	DIPLEX ULTRACONE
	ZENITH NOISE TEMP (DEG K)	29.08	-1.00	+3.00	
	ZENITH NOISE SPEC DEN (DBM/HZ)	-183.98	-0.15	+0.43	
13	CARRIER POWER/TOTAL POWER (DB)	-2.44	0.41	-1.30	
14	RECEIVED CARRIER POWER (DBM) (11+13)	-161.47	+2.41	-3.30	
15	CARRIER THRESHOLD NOISE BW (DB.HZ)	10.80	-1.00	0.00	
	CARRIER TRACKING (ONE-WAY)				
16	THRESHOLD SNR IN $2B_{LO}$ (DB)	0.00	0.00	0.00	
17	THRESHOLD CARRIER POWER (DBM) (12+15+16)	-172.36	-2.00	+0.36	
18	PERFORMANCE MARGIN (DB) (14-17)	10.89	+4.41	-3.66	
	DATA CHANNEL (8-1/3)				
19	DATA POWER/TOTAL POWER (DB)	-3.66	+1.27	-0.61	θ 41.0$^{+8.4}_{-3.3}$ DEG
20	WAVEFORM DISTORTION LOSS (DB)	0.00	0.00	0.00	
21	LOSS THROUGH RADIO SYSTEM (DB)	-1.95	+0.20	-0.20	
	UPLINK SNR IN $2B_{LO}$ (DB) 1-WAY	—	—	—	
	DOWNLINK SNR IN $2B_{LO}$ (DB)	—	—	—	
22	SUBCARRIER DEMOD LOSS (DB)	-0.40	+0.10	-0.10	
23	BIT SYNC/DETECTION LOSS (DB)	-0.10	+0.10	-0.10	
24	RECEIVED DATA POWER (DBM) (11+19+20+21+22+23)	-165.14	+3.67	-3.01	
25	THRESHOLD DATA POWER (DBM) (12+25a+25b)	-168.75	-0.13	+0.36	
a	THRESHOLD PT/N (DB) (BER = 5 IN 10^3)	5.20	0.00	0.00	
b	BIT RATE (DB.BPS)	9.21	0.00	0.00	
26	PERFORMANCE MARGIN (DB) (24-25)	3.61	+3.80	-3.37	
	DATA CHANNEL (33-1/3)				
27	DATA POWER/TOTAL POWER (DB)	-3.66	+1.27	-0.61	θ 41.0$^{+8.4}_{-3.3}$ DEG
28	WAVEFORM DISTORTION LOSS (DB)	0.00	0.00	0.00	
29	LOSS THROUGH RADIO SYSTEM (DB)	-0.75	0.10	-0.10	
	UPLINK SNR IN $2B_{LO}$ (DB) 1-WAY	—	—	—	
	DOWNLINK SNR IN $2B_{LO}$ (DB)	—	—	—	
30	SUBCARRIER DEMODULATION LOSS (DB)	-0.25	+0.10	-0.10	
31	BIT SYNC/DETECTION LOSS (DB)	-0.10	+0.10	-0.10	
32	RECEIVED DATA POWER (DBM) (11+27+28+29+30+31)	-163.79	+3.57	-2.91	
33	THRESHOLD DATA POWER (DBM) (12+33a+33b)	-162.73	-0.13	+0.36	
a	THRESHOLD PT/N (DB) (BER = 5 IN 10^3)	5.20	0.00	0.00	
b	BIT RATE (DB.BPS)	15.23	0.00	0.00	
34	PERFORMANCE MARGIN (DB) (32-33)	-1.06	+3.70	-3.27	

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TABLE 7H-3. PLAYBACK MODE 6 HIGH RATE TELEMETRY

TELEMETRY DESIGN CONTROL TABLE

TELEMETRY MODE: HIGH RATE PLAYBACK DATE: 1-31-67

SCIENCE CHANNEL: 8.1 KBPS ENGINEERING CHANNEL: 8-1/3 BPS STATION:

210 FOOT TRACKING STATION; (5 FOOT) ANTENNA TRACKING EARTH

LAUNCH: ARRIVAL:

TIME IN MISSION: DATE 7-14-74 HOURS FROM INJECTION:

NO.	PARAMETER	NOMINAL VALUE	TOLERANCE FAVORABLE	TOLERANCE ADVERSE	
1	TOTAL TRANSMITTER POWER (DBM)	42.32	+0.50	-0.50	
2	TRANSMITTING CIRCUIT LOSS (DB)	-1.45	+0.25	-0.25	
3	TRANSMITTING ANTENNA GAIN (DB)	28.70	+0.32	-0.32	
4	TRANSMITTING ANTENNA POINTING LOSS (DB)	0.00	0.00	0.00	
5	SPACE LOSS (DB) F = 2297 MHz, R = 368.35 x 10^6 KM	-271.00	0.00	0.00	
6	POLARIZATION LOSS (DB)	-0.03	+0.02	-0.02	
7	RECEIVING ANTENNA GAIN (DB)	61.40	+0.40	-0.40	
8	RECEIVING ANTENNA POINTING LOSS (DB)	0.00	0.00	0.00	
9	RECEIVING CIRCUIT LOSS (DB)	0.00	0.00	0.00	
10	NET CIRCUIT LOSS (DB) (+3+4+5+6+7+8+9)	-182.38	+0.99	-0.99	
11	TOTAL RECEIVED POWER P(T) (DBM) (1+10)	-140.06	+1.49	-1.49	
12	RECEIVER NOISE SPECTRAL DENSITY (DBM/HZ) ELEV = 25 DEG	-183.16	-0.13	+0.36	
	ZENITH NOISE TEMP (DEG K)	29.00	-1.00	+3.00	DIPLEX
	ZENITH NOISE SPEC DEN (DBM/HZ)	-183.98	-0.15	+0.43	ULTRACONE
13	CARRIER POWER/TOTAL POWER (DB)	-7.67	+1.57	-2.05	
14	RECEIVED CARRIER POWER (DEM) (11+13)	-147.73	+3.06	-3.54	
15	CARRIER THRESHOLD NOISE BW (DB. HZ)	10.80	-1.00	+0.00	
	CARRIER TRACKING (ONE-WAY)				
16	THRESHOLD SNR IN $2B_{LO}$ (DB)	0.00	0.00	0.00	
17	THRESHOLD CARRIER POWER (DBM) (12+15+16)	-172.36	-1.13	+0.36	
18	PERFORMANCE MARGIN (DB) (14-17)	24.63	+4.18	-3.89	
	DATA CHANNEL (8100)				
19	DATA POWER/TOTAL POWER (DB)	-1.04	+0.31	-0.39	θ 65 $^{+5.4}_{-5.3}$ DEG
20	WAVEFORM DISTORTION LOSS (DB)	-0.30	0.00	0.00	
21	LOSS THROUGH RADIO SYSTEM (DB)	-0.10	0.00	-0.10	
	UPLINK SNR IN $2B_{LO}$ (DB) 1-WAY	—	—	—	
	DOWNLINK SNR IN $2B_{LO}$ (DB)	—	—	—	
22	SUBCARRIER DEMOD LOSS (DB)	-0.10	0.00	-0.10	
23	BIT SYNC/DETECTION LOSS (DB)	-0.20	0.00	-0.10	
24	RECEIVED DATA POWER (DBM) (11+19+20+21+22+23)	-141.80	+1.80	-2.18	
25	THRESHOLD DATA POWER (DBM) (12+25a+25b)	-141.07	-0.13	+0.36	
a	THRESHOLD PT/N (DB) (WER = 1 IN 10^2)	3.00	0.00	0.00	
b	BIT RATE (DB. BPS)	39.09	0.00	0.00	
26	PERFORMANCE MARGIN (DB) (24-25)	-0.73	+1.92	-2.53	
	DATA CHANNEL (8-1/3)				
27	DATA POWER/TOTAL POWER (DB)	-21.20	+0.96	-1.51	θ 11.9 $^{+1.5}_{-1.4}$ DEG
28	WAVEFORM DISTORTION LOSS (DB)	0.00	0.00	0.00	
29	LOSS THROUGH RADIO SYSTEM (DB)	-0.80	+0.20	-0.20	
	UPLINK SNR IN $2B_{LO}$ (DB) 1-WAY	—	—	—	
	DOWNLINK SNR IN $2B_{LO}$ (DB)	—	—	—	
30	SUBCARRIER DEMODULATION LOSS (DB)	-0.40	+0.10	-0.10	
31	BIT SYNC/DETECTION LOSS (DB)	-0.10	+0.10	-0.10	
32	RECEIVED DATA POWER (DBM) (11+27+28+29+30+31)	-162.56	2.85	-3.40	
33	THRESHOLD DATA POWER (DBM) (12+33a+33b)	-168.75	-0.13	+0.36	
a	THRESHOLD PT/N (DB) (BER = 5 IN 10^3)	5.20	0.00	0.00	
b	BIT RATE (DB. BPS)	9.21	0.00	0.00	
34	PERFORMANCE MARGIN (DB) (32-33)	6.19	2.97	-3.75	

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FIG. 7H-5

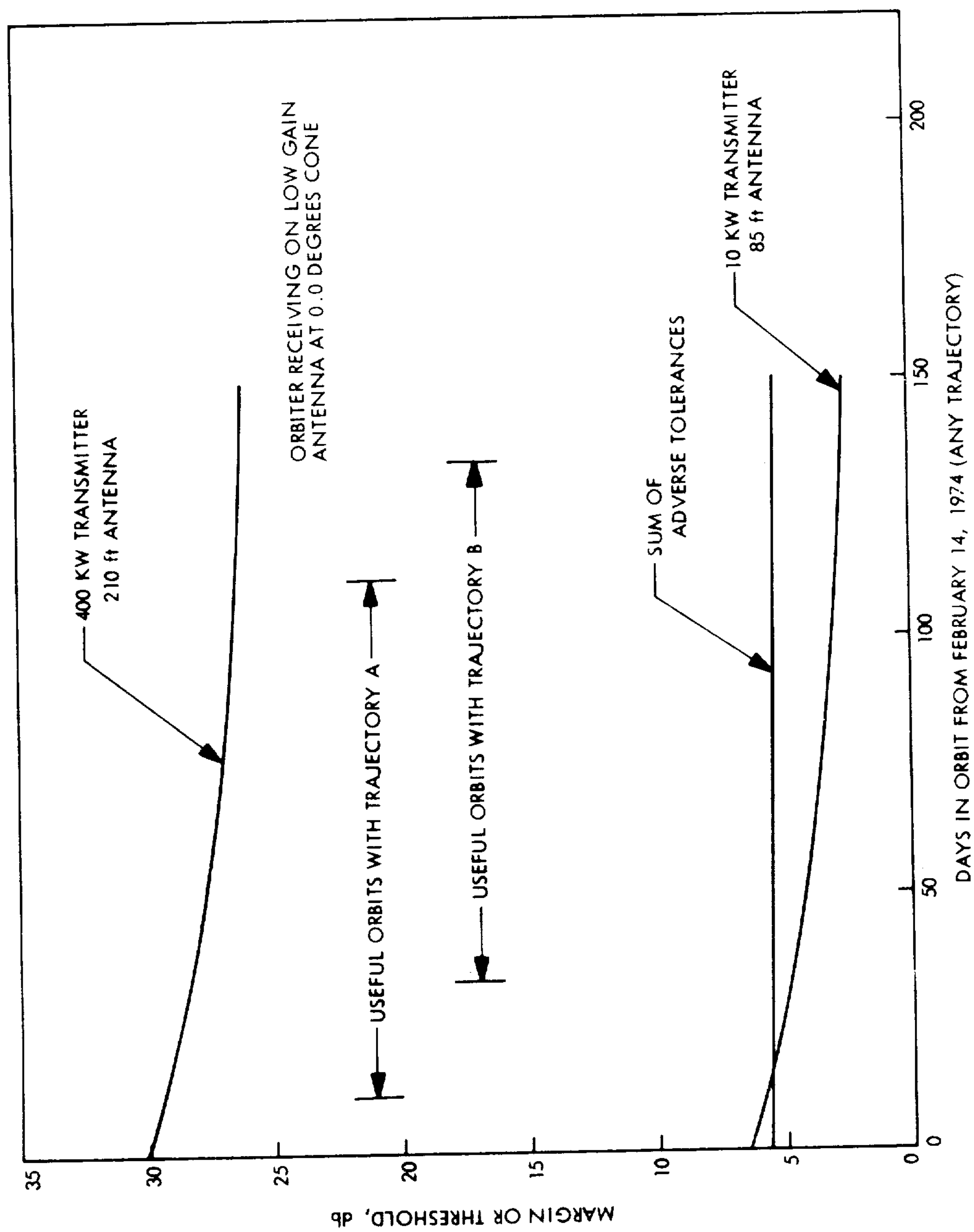

Figure 7H-5. Command Performance During Orbit

TABLE 7H-4. COMMAND CAPABILITY NEAR END OF MISSION

COMMAND DESIGN CONTROL TABLE

COMMAND MODE: FINAL COMMANDS DATE: 1-31-69

DATA CHANNEL: 1 BPS SYNC CHANNEL: 1 BPS STATION:

210 FOOT TRACKING STATION; LOW GAIN ANTENNA AT 0.0 CONE

LAUNCH: ARRIVAL:

TIME IN MISSION: DATE 7-14-74 HOURS FROM INJECTION:

NO.	PARAMETER	NOMINAL VALUE	TOLERANCE FAVORABLE	TOLERANCE ADVERSE
1	TOTAL TRANSMITTER POWER (DBM)	86.00	+0.50	-0.00
2	TRANSMITTING CIRCUIT LOSS (DB)	0.00	0.00	0.00
3	TRANSMITTING ANTENNA GAIN (DB)	59.50	+0.60	-0.60
4	TRANSMITTING ANTENNA POINTING LOSS (DB)	0.00	0.00	0.00
5	SPACE LOSS (DB) F = 2116 MHz, R = 368.35 x 10^6 KM	-270.29	0.00	0.00
6	POLARIZATION LOSS (DB)	-0.04	+0.02	-0.02
7	RECEIVING ANTENNA GAIN (DB)	7.00	+0.40	-0.40
8	RECEIVING ANTENNA POINTING LOSS (DB)	-0.48	+0.08	-0.08
9	RECEIVING CIRCUIT LOSS (DB)	-0.11	+0.01	-0.01
10	NET CIRCUIT LOSS (DB) (2+3+4+5+6+7+8+9)	-204.41	+1.11	-1.11
11	TOTAL RECEIVED POWER P(T) (DBM) (1+10)	-118.41	+1.61	-1.11
12	RECEIVER NOISE SPECTRAL DENSITY (DBM/HZ) ELEV = DEG ZENITH NOISE TEMP (DEG K) ZENITH NOISE SPEC DEN (DBM/HZ)	-167.34	-1.34	+1.62
13	CARRIER POWER/TOTAL POWER (DB)	-2.50	+0.25	-0.25
14	RECEIVED CARRIER POWER (DBM) (11+13)	-120.91	+1.86	-1.36
15	CARRIER THRESHOLD NOISE BW (DB.HZ) $14^{-2.5}_{+0.5}$	11.46	-0.85	+0.15
	CARRIER TRACKING (ONE-WAY)			
16	THRESHOLD SNR IN $2B_{LO}$ (DB)	0.00	0.00	0.00
17	THRESHOLD CARRIER POWER (DBM) (12+15+16)	-155.88	-2.19	1.77
18	PERFORMANCE MARGIN (DB) (14-17)	34.97	4.06	-3.14
	DATA CHANNEL			
19	DATA POWER/TOTAL POWER (DB)	-9.48	+0.39	-0.78
20	WAVEFORM DISTORTION LOSS (DB)	0.00	0.00	0.00
21	LOSS THROUGH RADIO SYSTEM (DB)	-1.50	+0.20	-0.20
	UPLINK SNR IN $2B_{LO}$ (DB) 1-WAY	—	—	—
	DOWNLINK SNR IN $2B_{LO}$ (DB)	—	—	—
22	SUBCARRIER DEMOD LOSS (DB)	0.00	0.00	0.00
23	BIT SYNC/DETECTION LOSS (DB)	0.00	0.00	0.00
24	RECEIVED DATA POWER (DBM) (11+19+20+21+22+23)	-129.39	2.20	-2.09
25	THRESHOLD SYNC POWER (DBM) (12+25a+25b)	155.64		
a	THRESHOLD PT/N (DB) (BER = 1 IN 10^5)	11.70	-1.00	+1.00
b	BIT RATE (DB.BPS)	0.00	0.00	0.00
26	PERFORMANCE MARGIN (DB) (24-25)	26.25	3.20	-3.09
	SYNC CHANNEL			
27	SYNC POWER/TOTAL POWER (DB)	-5.73	+0.48	-0.89
28	WAVEFORM DISTORTION LOSS (DB)	0.00	0.00	0.00
29	LOSS THROUGH RADIO SYSTEM (DB)	-1.50	+0.20	-0.20
	UPLINK SNR IN $2B_{LO}$ (DB) 1-WAY	—	—	—
	DOWNLINK SNR IN $2B_{LO}$ (DB)	—	—	—
30	SUBCARRIER DEMODULATION LOSS (DB)	0.00	0.00	0.00
31	BIT SYNC/DETECTION LOSS (DB)	0.00	0.00	0.00
32	RECEIVED SYNC POWER (DBM) (11+27+28+29+30+31)	-125.64	2.29	-2.20
33	THRESHOLD SYNC POWER (DBM) (12+33a+33b)	-152.64	-2.34	2.62
a	THRESHOLD PT/N (DB) (BER = 1 IN 10^5)	14.70	-1.00	+1.00
b	BIT RATE (DB.BPS)	0.00	0.00	0.00
34	PERFORMANCE MARGIN (DB) (32-33)	27.00	4.63	-4.82

T_{ANT} 43 ± 5°K

NF_{RCVR} 7.4 $^{+1.4}_{-1.1}$ db

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d. Relay Link Before Lander Touchdown

During the flight when the lander capsule is attached to the spacecraft, a multiconductor cable connects the ends of the relay link together. There are no transmitters or receivers required by the relay link at this time.

When the landing capsule is released, its 400 MHz transmitter must immediately begin sending data to the orbiter. Signal levels will be abnormally strong at first because the receiver is so close to the transmitter. However, the bioshield base will be hiding the relay link antenna on the orbiter. It is possible that a small probe on the cover will allow enough signal to leak through to provide reliable communication. This probe, which can be later jettisoned with the bioshield base, may have to act more like an attenuator than like a normal antenna.

e. Relay Link Turn-on Command Beacon

The beacon on the orbiter which transmits the turn-on command operates only intermittently. It begins operation after the lander is on the ground. An AM modulated command will be transmitted to the lander approximately every minute. When the lander is sufficiently close, and signal levels are sufficiently strong, stored telemetry data will be transmitted from the lander to the orbiter. More work remains to be done on the relay link beacon to determine the exact power levels required for reliable operation.

f. Relay Link Telemetry

The performance margin of the entry relay link for the out-of-orbit delivery mode has not been calculated. Preliminary investigations of possible spread of look angles indicate that it may be possible to use one antenna array to provide coverage of entry and landed phase for the majority of orbits considered.

In order to analyze the relay links after capsule touchdown for various orbits as well as optimize the location of each relay antenna, a figure-of-merit (FOM) function was utilized as follows:

$$FOM(t) = \frac{G_{SC} \cdot G_L}{R^2(t)} \quad (1)$$

where G_{SC} is the spacecraft receiver antenna effective gain
G_L is the lander/capsule transmitter antenna effective gain
$R(t)$ is the lander/capsule-to-spacecraft range

The effective gain of each antenna was approximated using the equation,

$$G = G_{peak}\left(\frac{\sin F}{F}\right)^2 \quad (2)$$

$$\text{where} \quad F = 2.78\,\frac{\Theta(t)}{\Theta_{hp}} \quad ; \quad F(t) \leq \pi \quad (3)$$

$\Theta(t)$ = angle off antenna axis

Θ_{hp} = antenna half-power beamwidth

In order to isolate the effect of the trajectory geometry on the relay link FOM, equation (1) was separated into two parts as follows:

$$FOM = \left[\frac{G_{s/c\,peak} \cdot G_{L\,peak}}{R^2_{max}}\right]\left[\frac{R^2_{max}}{R^2(t)}\left(\frac{\sin F_{s/c}}{F_{s/c}}\right)^2\left(\frac{\sin F1}{F_1}\right)^2\right] \quad (4)$$

The second term was then used to represent the effects of trajectory geometry. For convenience, the second term was converted to decibels as follows:

$$db_{margin} = 10\log_{10}\left[\frac{R^2\,Max}{R^2(t)}\left(\frac{\sin F_{s/c}}{F_{s/c}}\right)^2\left(\frac{\sin F_1}{F_1}\right)^2\right] \quad (5)$$

If design control tables are referenced to 0.0 db antenna pointing losses and maximum communication range, a positive db margin for equation (5) represents positive performance margin for the link. Figure 7H-6 shows a plot of decibel margin for a typical lander/orbiter relay link. The plot for trajectory A with link parameters as shown in Table 7H-5 covers the initial pass and the ninety day pass. The program used assumes a spherical planet and does not take into account effects due to oblateness.

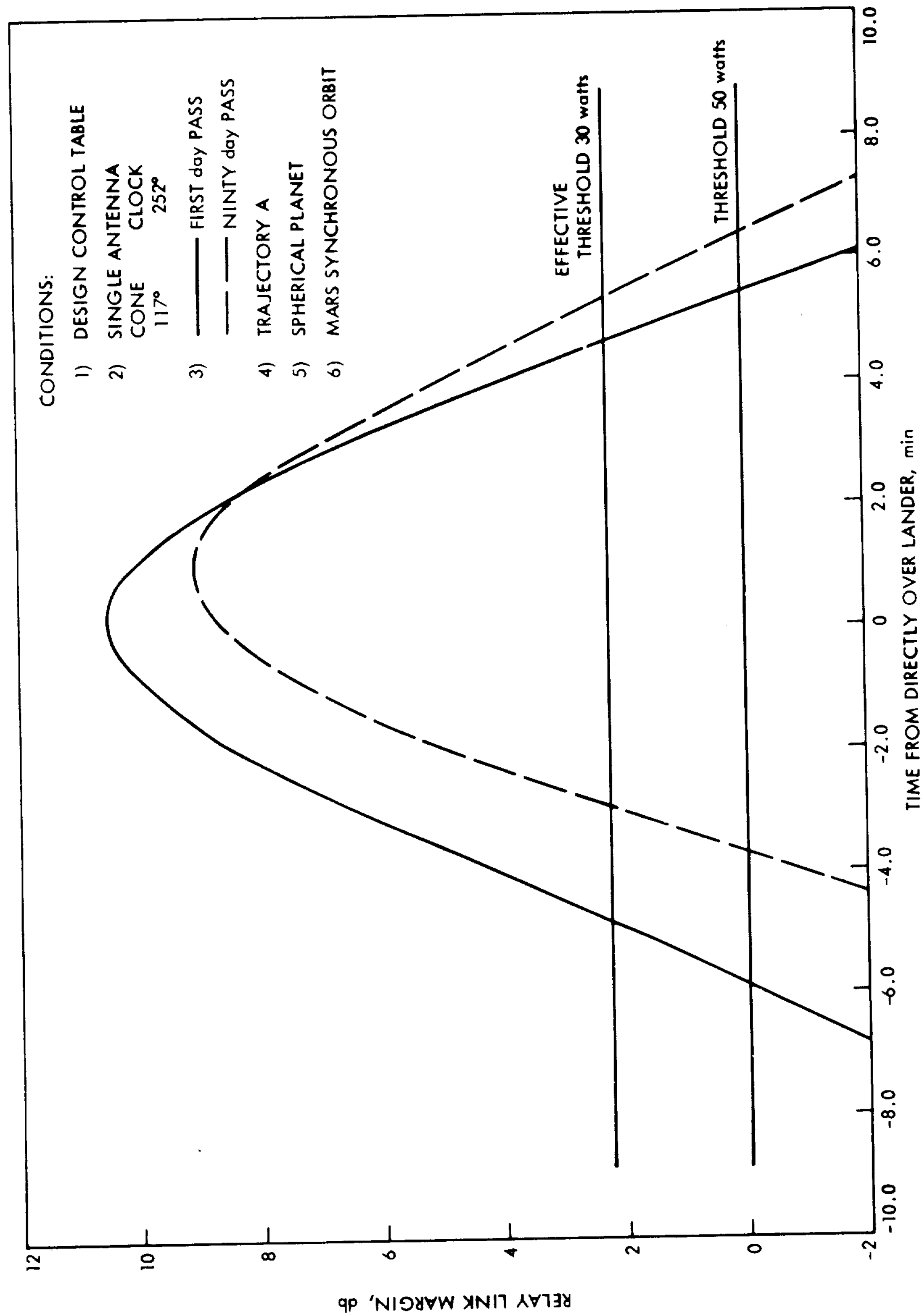

Figure 7H-6. Lander Relay Performance -- Trajectory B

Table 7H-5. Viking Orbiter Post Landed - Lander to Orbiter Link

No.	PARAMETER	VALUE dbm	TOLERANCE		SOURCE
			FAV.	ADVERSE	
1	TOTAL TRANSMITTER POWER 50w	47	0.0	-1.0	
2	TRANSMITTING CIRCUIT LOSS	-1.5	0.2	-0.4	
3	TRANSMITTING ANTENNA GAIN	5	1.0	-1.0	
4	TRANSMITTING ANTENNA POINTING LOSS		0	-0.75	
5	SPACE LOSS @ 400 mc, $R = 3.5 \times 10^3$ km	-155.4	-	-	
6	POLARIZATION LOSS	0.0	0.7	-0.7	
7	RECEIVING ANTENNA GAIN	5	1.0	-1.0	
8	RECEIVING ANTENNA POINTING LOSS		0	-0.75	
9	RECEIVING CIRCUIT LOSS	-1.5	0.4	-0.4	
10	RECEIVER LIMITING LOSS	-1.0	+2.0	-0.2	
11	MULTIPATH LOSSES	-1.5	0.9	-1.7	
12	SYNC LOSSES	-0.5	+0.2	-0.2	
13	NET CIRCUIT LOSS	-151.4	6.4	-7.1	
14	TOTAL RECEIVED POWER	-104.4	6.4	-8.1	
15	RECEIVER NOISE SPECTRAL DENSITY (N/B) $T_S = 705 {}^{+180}_{-145}$ °K ·	-170.1	-1.0	+1.0	
16	BIT PERIOD 20K bps $\pm$5% NRz	43	-0.2	+0.2	
17	RECEIVED ST/N/B	22.7	+7.6	-9.3	
18	REQUIRED ST/N/B for $P_E{}^B = 10^{-3}$ N' = 3	12.2	-1.0	+1.0	
19	PERFORMANCE MARGIN	10.5	+8.6	-10.3	

The preceding analysis assumes the use of a symmetrical antenna with approximately 5-db on axis gain and a 104 degree half-power beamwidth. The antenna axis is aligned along a cone angle of 117 degrees and a clock angle of 252 degrees.

Preliminary examination of the landed view angles and entry view angles indicates that positive performance margin must be maintained over the solid angle defined by an antenna mounted with the axis aligned along a cone angle of 138° and a clock angle of 259° with the semi-major axis of the antenna platform aligned with the plane containing the spacecraft-to-lander vector at 80 cone, 220 clock, and 150 cone, 360 clock. Further effort is required to match the antenna characteristics to the range of possible orbits and then determine the performance of the link.

SECTION VIII

BASELINE ORBITER SUBSYSTEM DESCRIPTIONS

A. STRUCTURE

1. Introduction

The structural design of the Viking orbiter is based on the technology developed for the Mariner series spacecraft. The Viking has requirements for much greater capability than the Mariner '71 and this results in a completely new structural design.

Figure 7B-1 shows the configuration for the orbiter. Each major structural component will be discussed. A weight estimate for each is shown in Table 8A-1. Also shown are estimates for the lander capsule adapter and the S/C adapter weights chargeable to the orbiter system weight allocation. These weights are obviously dependent on the criteria, loads and assumptions used in the preliminary analysis. These are discussed in the appendix.

2. Preliminary Structural Analysis

The orbiter system was modeled using a rigid capsule and assumed launch vehicle stub adapter stiffnesses to permit a structural analysis to be made from which dynamic characteristics and structural member sizes and weights could be estimated. All members were assumed to be made of aluminum.

The structural elements were sized on the basis of not exceeding 30,000 psi when subjected to a 1.0g (0-peak) lateral sinusoidal acceleration at the base of the S/C adapter with shear at the base limited to 4 times the weight of the total system but not limited by base bending moment. A static loading of 10 g axial or 5 g lateral was checked and found not to be critical on any members used in this analysis.

A finite element analysis was performed to verify the design and to estimate structural weights. The analytical model was doubly symmetric

Table 8A-1. Structure Subsystem Weight Estimate

		Weight (lbs)
Structure		
Octogon		74.0
Chassis Subassemblies (12)		68.0
Cable Support Structure		4.0
Solar Panel Structure		89.0
Substrates (115 sq. ft.)	78.0	
Outriggers	11.0	
Misc. Bracketry and Attachments		8.0
Antenna Support Structure		12.5
High Gain	6.0	
Low Gain	1.5	
Relay Antenna Bracket	2.0	
(Solar Panel Beef-up)	3.0	
Meteoroid Shield		14.0
Fasteners		15.0
(Propulsion Module Structure is included in Propulsion Weight)		
Total		284.5
Lander Capsule Adapter		
Structure (truss)		25.0
Separation Assembly		10.0
Harness Assembly		7.0
Total		42.0
Spacecraft Adapter		
Structure		122.0
Separation Assembly		15.0
Wiring Harness		8.5
Inflight Disconnect		3.5
Environmental Instrumentation		6.0
Other equipment in adapter see Section V.F.		29.5
Total		184.5

and utilized 6 rigid components interconnected by structural elements. Each of these components,

1) 1/4 lander capsule assembly,
2) 1/2 oxidizer tank,
3) 1/2 fuel tank,
4) 1/4 pressure tank and motor assembly,
5) 1/4 planet scan platform, and
6) solar panel

has 6 degrees of freedom with appropriate mass coupling for the solar panel. In addition, the 1/4 bus has 3 mass points with only uncoupled translational mass. The L/V adapter interface was assumed to provide an uncoupled axial stiffness of 8×10^5 lb/in, tangential stiffness of 3×10^5 lb/in and radial stiffness of 5×10^3 lb/in at each of 12 truss points. Each mode was assumed to have 1.0% critical damping unless the 4 g shear was exceeded.

The last iteration of the analysis showed about as many members over the nominal stress as under and all in the 25,000 to 40,000 psi range. Only the spacecraft adapter top ring was updated in weight at this iteration. The solar panel was given an additional rigid body degree of freedom about its long axis causing its support to be statically determinant to prevent it carrying composite loads. All other rigid bodies acted as large structural elements of the composite. A check was made to ensure that elements with the calculated areas can have sufficient diameters to prevent buckling without exceeding crippling stresses.

A tabulation of the natural frequencies of the structural model is given in Table 8A-2.

3. Orbiter Bus

The orbiter bus is designed using the structural concepts of MM71 though it supports quite different loads. It consists of substantial upper and lower rings with short longerons between them. Shear capability is provided by the electronic subassemblies when they are bolted in place.

The spacecraft is attached to the spacecraft adapter at the four longerons in the middle of the long bays. The attachment is made to the adapter through a pyrotechnically separable bolt assembly.

The propulsion module supports are ties to the bus through the ring at these four points, thus transferring loads to the adapter in a direct load path.

Table 8A-2. Orbiter Structural Model Modal Frequencies

Launch Configuration

Frequency, cps	Symmetry*	Limitation Criteria
7.4	SS	acceleration input (acc.)
22.	SS	acc.
30.	SS	force
36.	SS	force
43.	SS	acc.
50.	SS	acc.
6.4	SA	force
12.	SA	acc.
16.	SA	acc.
17.	SA	acc.
25.	SA	acc.
34.	SA	acc.
6.5	AS	force
12.	AS	acc.
16.	AS	acc.
18.	AS	acc.
31.	AS	acc.
35.	AS	acc.
15.	AA	--
16.	AA	--
19.	AA	--
33.	AA	--
43.	AA	--
52.	AA	--

*SS - Symmetric Symmetric--includes axial mode.

SA - Symmetric Anti-Symmetric--includes bending in XZ plane.

AS - Anti-Symmetric Symmetric--includes bending in YZ plane.

AA - Anti-Symmetric Anti-Symmetric--includes torsion.

The lander capsule adapter is attached to the bus by a truss at the four longeron points. The truss supports for the centerline mounted scan platform tie in at these points also. The ring at this end of the bus is sized primarily by the loads from these components.

Outriggers are bolted to the longerons at the short sides of the bus -- similar to MM71 -- to provide support for the solar panels at their hinge line. Cable harnesses and attitude control plumbing are routed on the outriggers to get to the solar panels.

The high gain antenna is stowed on the side of the bus and is supported by a truss structure which is deployable to place the antenna in its initial pointing position. Pointing updates are made by rotating the antenna about the deployed hinge axis with a stepper actuator.

4. Electronic Packaging

The configuration provides for twelve electronic assemblies of the type used on MM69 and MM71. Electronic assemblies, shown in Figures 8A-1 and 8A-2, are typical of the MM71 type and only minor modifications, if any, to the basic chassis design will be required for Viking. The batteries may require new chassis design depending on the type and size if they are different from the one selected for MM71. Table 8A-3 is a description of the location and function of the electronic assembly chassis. The only space for growth in the electronic assemblies as presently proposed would be in EA III. Approximately 80% of that assembly is available.

Electronic sensors and experiments requiring defined fields of view are located outside of the electronic assembly compartment on the structure, solar panels or scan platform.

5. Solar Panel Substrates

The solar panels for the Viking orbiter are 48" by 84" which gives a gross area of 115 sq. ft. (MM71 has 83 sq. ft.). The MM71 technology is applied to these panels. The cell surface is a single skin on transverse corrugations supported on beams. The panels are folded down and their tips are supported by the spacecraft adapter during launch. A damper assembly

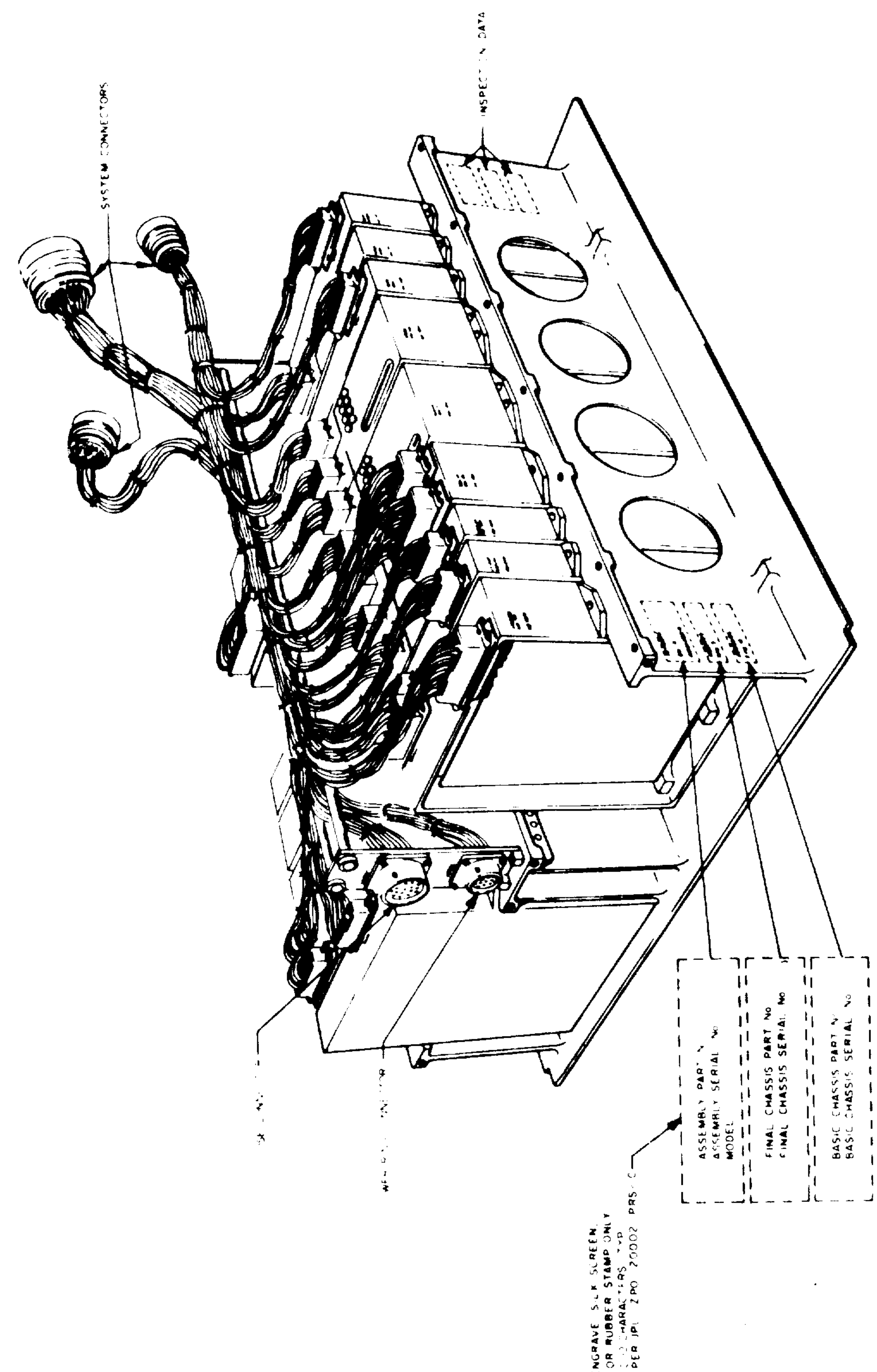

Figure 8A-1. Typical Type I Assembly

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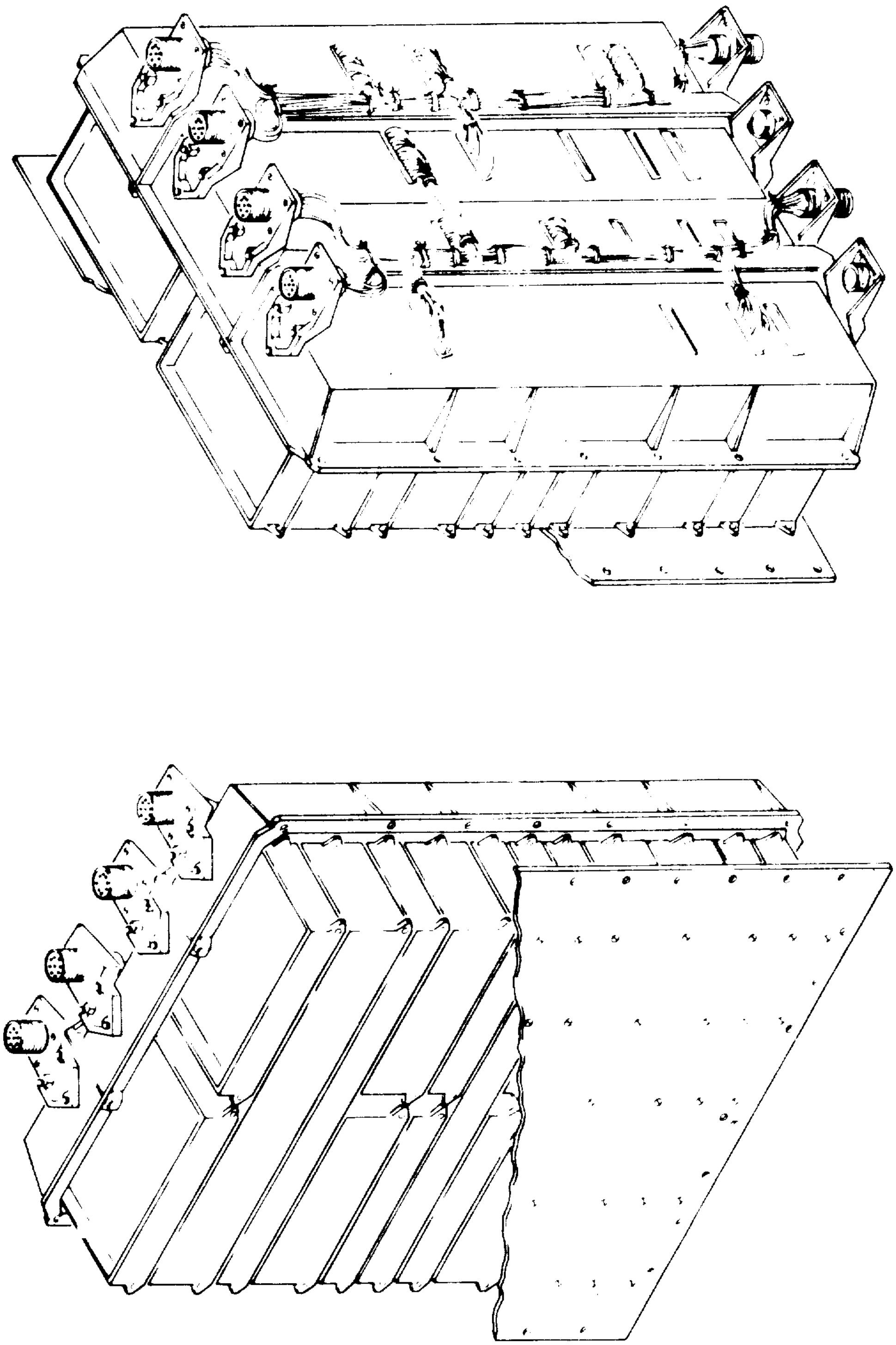


Figure 8A-2. Typical Type II Assembly

Table 8A-3. Electronic Assembly Identification

ELECTRONIC ASSEMBLY	FUNCTION
EA I[2]	DATA STORAGE
EA II A[1]	DSS/SPARE SPACE (80%)
EA II B[2]	COMMAND/TELEMETRY
EA III	ATTITUDE CONTROL/CC&S
EA IV A[2]	RELAY RADIO/RADIO FREQUENCY
EA IV B[1]	RADIO FREQUENCY
EA V[2]	POWER CONVERSION
EA VI A[1][2]	BATTERY NO. 1
EA VI B[2]	BATTERY NO. 2
EA VII[2]	POWER REGULATOR
EA VIII A[1]	SCIENCE
EA VIII B	SDS/TV
	TOTAL

(1) NEW ASSEMBLIES FOR VIKING '73

(2) REVISED EA FROM MM'71

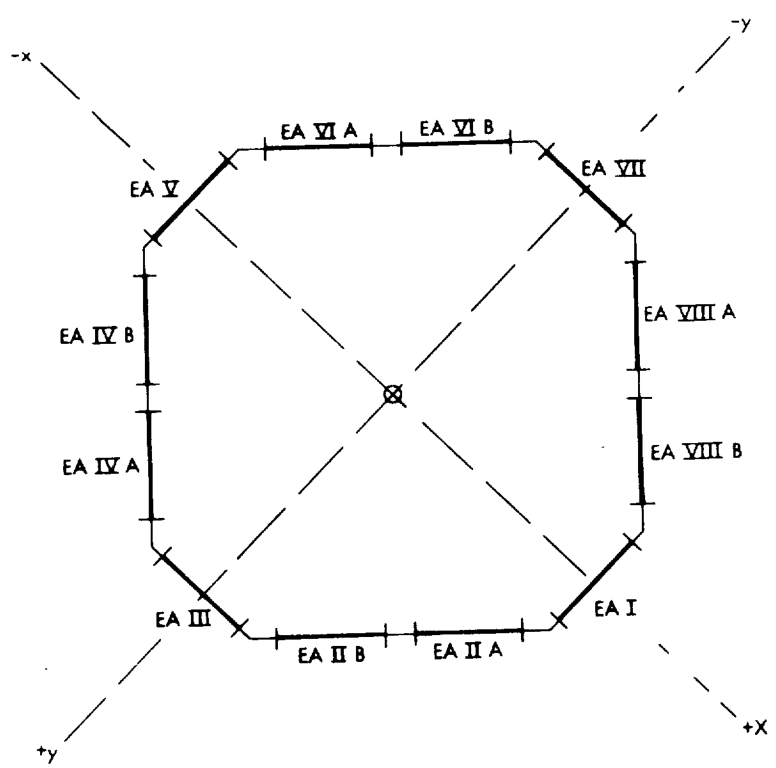
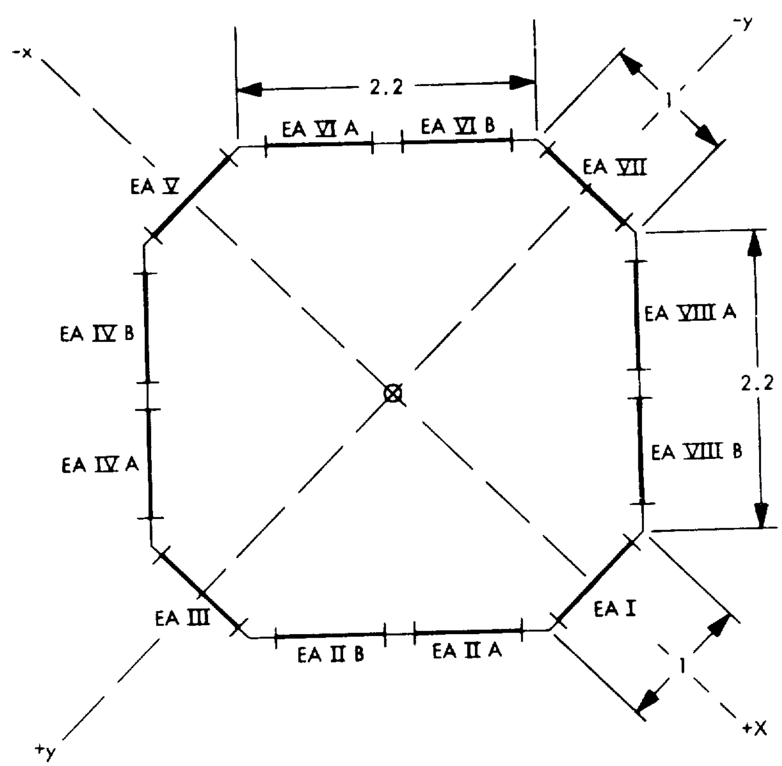


Table 8A-3. Electronic Assembly Identification

ELECTRONIC ASSEMBLY	FUNCTION
EA I(2)	DATA STORAGE
EA II A(1)	DSS/SPARE SPACE (80%)
EA II B(2)	COMMAND/TELEMETRY
EA III	ATTITUDE CONTROL/CC&S
EA IV A(2)	RELAY RADIO/RADIO FREQUENCY
EA IV B(1)	RADIO FREQUENCY
EA V(2)	POWER CONVERSION
EA VI A(1)(2)	BATTERY NO. 1
EA VI B(2)	BATTERY NO. 2
EA VII(2)	POWER REGULATOR
EA VIII A(1)	SCIENCE
EA VIII B	SDS/TV
	TOTAL

(1) NEW ASSEMBLIES FOR VIKING '73

(2) REVISED EA FROM MM'71



is required at this tiedown to attenuate vibration response of the panels.

The beams of one panel must support the relay antenna array. These beams require additional stiffness to do this.

6. Propulsion Module Structure

a. Propellant Tanks

The propellant tank design is based on the MM71 tank design. The major structural difference is that the four tanks, with 14-inch cylindrical sections added, are required for Viking. The tank support configuration requires changes in the tab locations on the tanks.

The material property data for the 6V 4 Al titanium which is being developed for MM71 will be directly applicable to the Viking tanks.

b. Pressure Vessels -- Pressurant Gas

Four pressure vessels, 18 inches in diameter are manifolded together and mounted on the end of the propellant tanks. These vessels are also made from 6V 4 Al titanium as are the two attitude control pressure vessels. The design and fabrication of these vessels will reflect the technology developed throughout the Mariner series spacecraft.

c. Support Structure

The propulsion module is designed to be an entity, with truss members tying the tanks, pressure vessels and thrust plate together. Tabs are fabricated on the tanks to accommodate the truss members attachment fittings. Motor alignment is made by adjustment of the motor on the thrust plate.

7. Meteoroid Protection

A meteoroid shield is required for the propulsion module tanks. The design proposed for MM71 incorporates a tightly woven fiberglass cloth, i.e., Armalon as the outer layer of the thermal insulation blanket and a thin (0.016)

24b

sheet of aluminum as a back up sheet. The tightly woven cloth is effective in breaking up the meteoroid particles and the back up sheet absorbs the energy of the residual gases and particles. The Armalon cloth is spaced at least 1/2" from the aluminum back up sheet.

Figure 8A-3 shows the effect on a particle which penetrated a sheet of teflon cover layer over a thermal insulation blanket. Figure 8A-4 shows the effect of a similar particle which penetrated a blanket with an outer layer of Armalon cloth at identical velocity conditions.

A meteoroid shield of this type of construction will ensure adequate probability of no catastrophic puncture of the propellant tanks.

8. Antenna

a. High Gain Reflector

The high gain antenna reflector is based on Mariner technology. It is 58" in diameter and is fabricated using aluminum honeycomb core sandwich with thin foil face sheets.

b. Low Gain Wave Guide

The low gain antenna wave guide is similar to those used on previous Mariners. The length and support are different for Viking because of the configuration changes. Its base is located as far outboard as possible on the bus and it is latched to the propulsion module during launch and deployed to a position parallel to the roll axis after spacecraft separation.

9. Lander Adapter

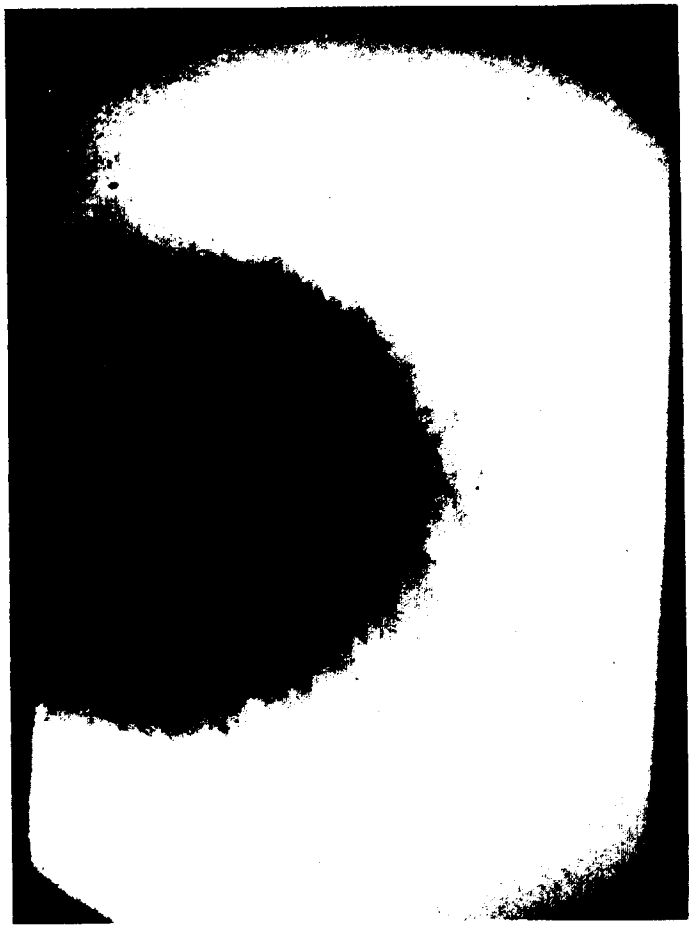
The lander adapter is an aluminum truss going from four separation points on the bus ring to eight at the capsule interface. The truss members are 2 1/2 inch diameter tubes. The capsule is assumed to have adequate rigidity to accommodate this configuration. A truss is not only desirable from a structural point of view; it is required to accommodate the strategy selected for landing site surveillance in which the instruments on the scan platform must

INSULATION PENETRATION DEBRIS

BACK-UP SHEET DAMAGE

COVER 0.001" TEFLON
BACK-UP SHEET 0.032" 6061-T6

Figure 8A-3. Meteoroid Penetration Test



INSULATION PENETRATION DEBRIS

BACK-UP SHEET DAMAGE

COVER BETA CLOTH
BACK-UP SHEET 0.063 6061-T6

Figure 8A-4. Meteoroid Protection Subsystem Test see out through the adapter.

Thermal leakage from the bus through the truss is taken to be small, even with aluminum, but a material with a lower conductivity may be required. The truss supports the lander umbilical harness which includes an RF line to the relay antenna required inside the canister. An in-flight disconnect is located at one of the truss points at the bus interface.

Separation assembly spring housings are located at each truss point also.

10. Spacecraft Adapter

The spacecraft adapter is also an aluminum truss structure spanning from the launch vehicle's 12-point truss adapter (station 2505.9 in.) to four separation points on the ring of the orbiter bus. The loads calculated for the vibration criteria discussed in section A-2 require 4-inch diameter tubes for the truss members. The diameter of these thin wall struts is controlled by a buckling criteria.

A large ring is located at the top of the spacecraft adapter and must work with the bus ring at that interface to react the kick loads at the orbiter-to-truss interface.

Values for the assumed launch vehicle stub adapter stiffness are given in section A-2. The stiffness is assumed to be available from the rings on the stub adapter and the thermal diaphragm support ring.

Miscellaneous equipment, including a wiring harness must be attached to the truss members. The location of concentrated masses will be such that bending loads in the members is minimized.

Specific attention will be given to the location of instrumentation to obtain flight loads data during launch.

Separation assembly spring housings are located at each truss point at the spacecraft interface.

B. RADIO FREQUENCY SUBSYSTEM

The radio frequency subsystem (RFS) receives uplink command and ranging signals transmitted by the Deep Space Instrumentation Facility (DSIF) and transmits telemetry and ranging signals to the DSIF. The RFS block diagram is shown in Figure 8B-1. Received and transmitted frequencies are in the S-band. When no uplink signal is being received, the transmitted frequency of 2295 MHz (nominal) is governed by an onboard crystal oscillator. When the receiver detects and achieves phase lock to an uplink signal, the transmitted frequency is 240/221 times the received frequency (2115 MHz nominal). When the spacecraft-transmitted frequency is under control of the onboard oscillator, the DSIF can accomplish one-way doppler tracking. When the spacecraft-transmitted frequency is under control of the uplink received frequency, the DSIF accomplishes two-way doppler tracking. The receiver and exciter portions of the RFS are functionally similar to those utilized for Mariner '71.

The spacecraft receiver recovers command and ranging information from the uplink signal. Command data is sent to the flight command subsystem. Ranging data, when present in the uplink signal, is sent through the RFS ranging channel to modulate the spacecraft transmitter in order to provide the turn-around ranging function. This ranging function can be turned on or off by means of ground command. A composite telemetry signal from the flight telemetry subsystem also modulates the spacecraft transmitter.

The receiver operates continuously and is a narrow band, automatic phase tracking double conversion superheterodyne. The receiver input stage is a balanced mixer, using hot carrier diodes. The spacecraft transmitter consists of the two exciters and two traveling wave tube (TWT) amplifiers. Either exciter may be used with either amplifier, providing a fully redundant transmitter capability. Switching between exciters and amplifiers is accomplished automatically by circuits which periodically test exciter and amplifier output. If the outputs are below a specified threshold, automatic transfer to the redundant equipment is accomplished. Switching may also be performed by ground command. Each of the amplifiers can be operated at 10- or 20-w nominal output. Power output change is by ground command and is accomplished by adjustment of operating voltages. Input drive is constant.

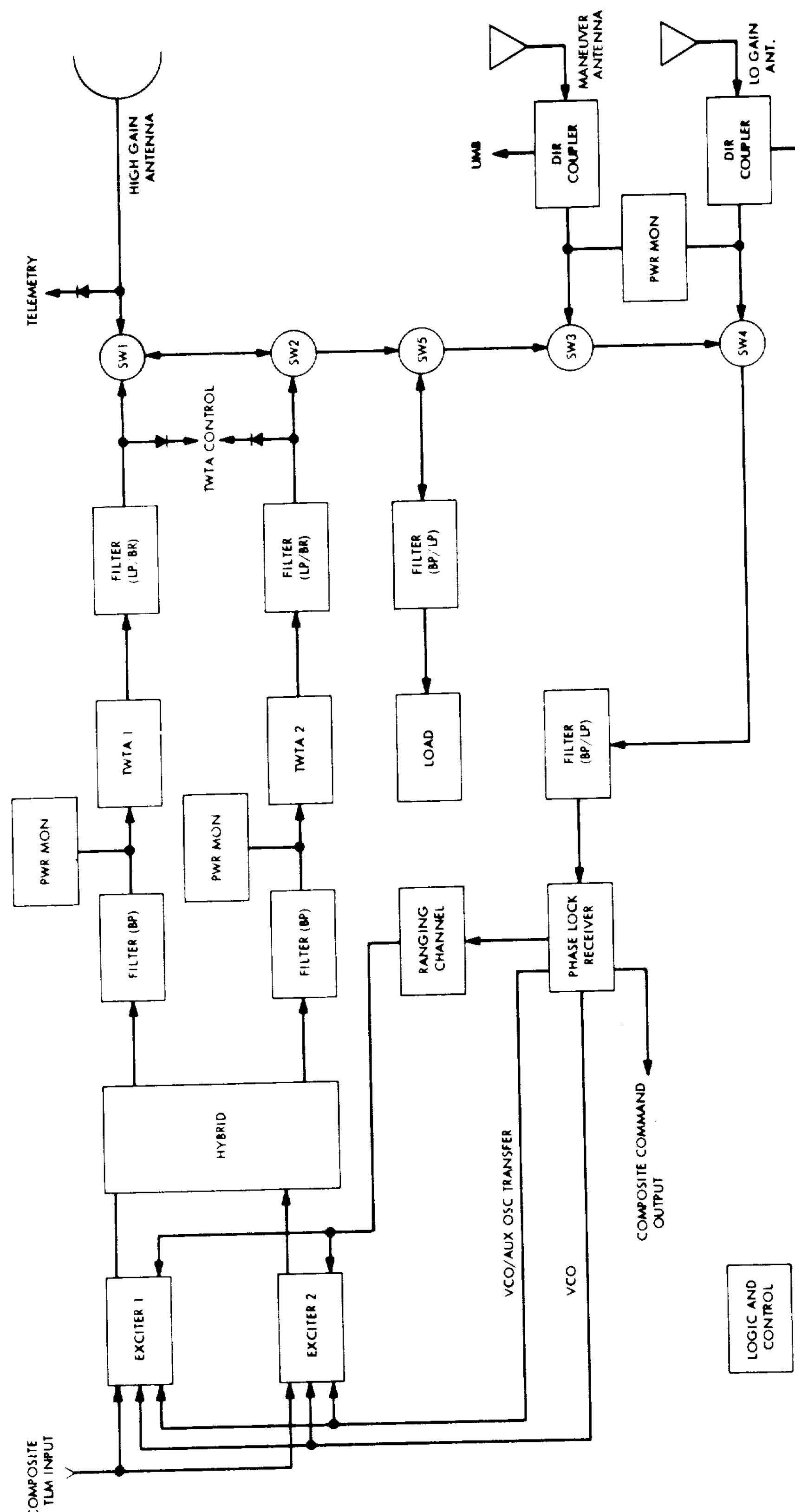


Figure 8B-1. Radio Frequency Subsystem Block Diagram

The RFS uses three antennas. The high-gain antenna, used for transmitting before and during orbit, employs a 58-in.-diameter parabolic reflector and a right-hand circularly polarized feed. One low-gain antenna, also circularly polarized, is mounted on the sunward side of the spacecraft and has a hemispherical pattern approximately symmetrical about the roll axis. It is used to receive and to transmit when the spacecraft is Sun oriented. The other low gain (maneuver) antenna is mounted in the -Z direction hemisphere and can be used during midcourse and orbit insertion maneuvers.

A summary of the radio frequency subsystem parameters is presented in Table 8B-1.

Table 8B-1. Summary of RFS Parameters

Uplink frequency	2115 MHz (nominal) Coherent PSK	
Downlink frequency	2295 MHz (nominal) Coherent PSK	
RF Output	10w/20w Dual mode	
Antennas:		
(1) High-gain	58" parabolic	Right hand circular polarized
(2) Low-gain		Right hand circular polarized
Transmitting circuit losses	1.5 db	

Weight and Power Requirements*

	Wt (lb)	Power (w)
High gain antenna	9.0	
Low gain antenna	5.0	
Maneuver antenna	3.0	
Transponder	17.0	23.0 (1)
Power amplifiers	20.0	99/62 (2)
Microwave components	11.0	
Control unit	3.4	1
Antenna feeds and cables	6.0	
Totals:	74.4	123/86

(1) One receiver and one exciter operating

(2) One PA: High power/Low power mode

*Note: The above estimates are based on internal R&D tasks.

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C. RELAY RADIO SUBSYSTEM

The relay radio subsystem (RRS) functions only after the Viking lander capsule separation begins. During capsule separation, cruise and entry, the RRS non-coherently receives and detects the 1348-bps frequency-shift-keyed (FSK) 400-MHz signal from the capsule and relays the information to the relay telemetry subsystem for subsequent transmission to Earth.

During the initial landing and three following orbital passes, the RRS non-coherently receives and detects the 20-Kbps FSK 400-MHz signal from the capsule and sends the information to the relay telemetry subsystem.

Additionally, the RRS contains a 4-w beacon transmitter which transmits a 400 MHz AM signal to the capsule as an indication to start the lander radio transmission. The subsystem consists (Figure 8C-1) of two (2) antennas (one mounted in the bioshield base) with switching and duplexing to allow required coverage during all mission phases. The 400-MHz signal is amplified by a preamplifier and mixed with a 370-MHz L.O. signal; the difference signal is amplified by the 30-MHz IF amplifier which has sufficient bandwidth (200 KHz) to handle either data rate with all expected doppler shifts and oscillator shifts. The amplified signal is fed to two pairs of detectors. The first pair of detectors is used to detect the low-rate FSK signal and has filter bandwidths of 18 KHz with a center frequency separation of 110 KHz. The bandwidth is sufficiently wide to accommodate the expected doppler and oscillator uncertainties. The amplitude of the output of the filters are measured by envelope detectors and the difference signal provides a binary data stream plus noise which is sent to the relay telemetry subsystem for processing.

The second pair of detectors is used to detect the high-rate FSK signal and has filter bandwidths of 55 KHz with a center frequency separation of 110 KHz. Filter center frequencies have been selected to allow use of the same frequency and frequency deviation in the lander transmitter. The difference signal from the high-rate (20,000 bps) detectors is also sent to the relay telemetry subsystem for processing.

Table 8C-1 lists a summary of RRS parameters.

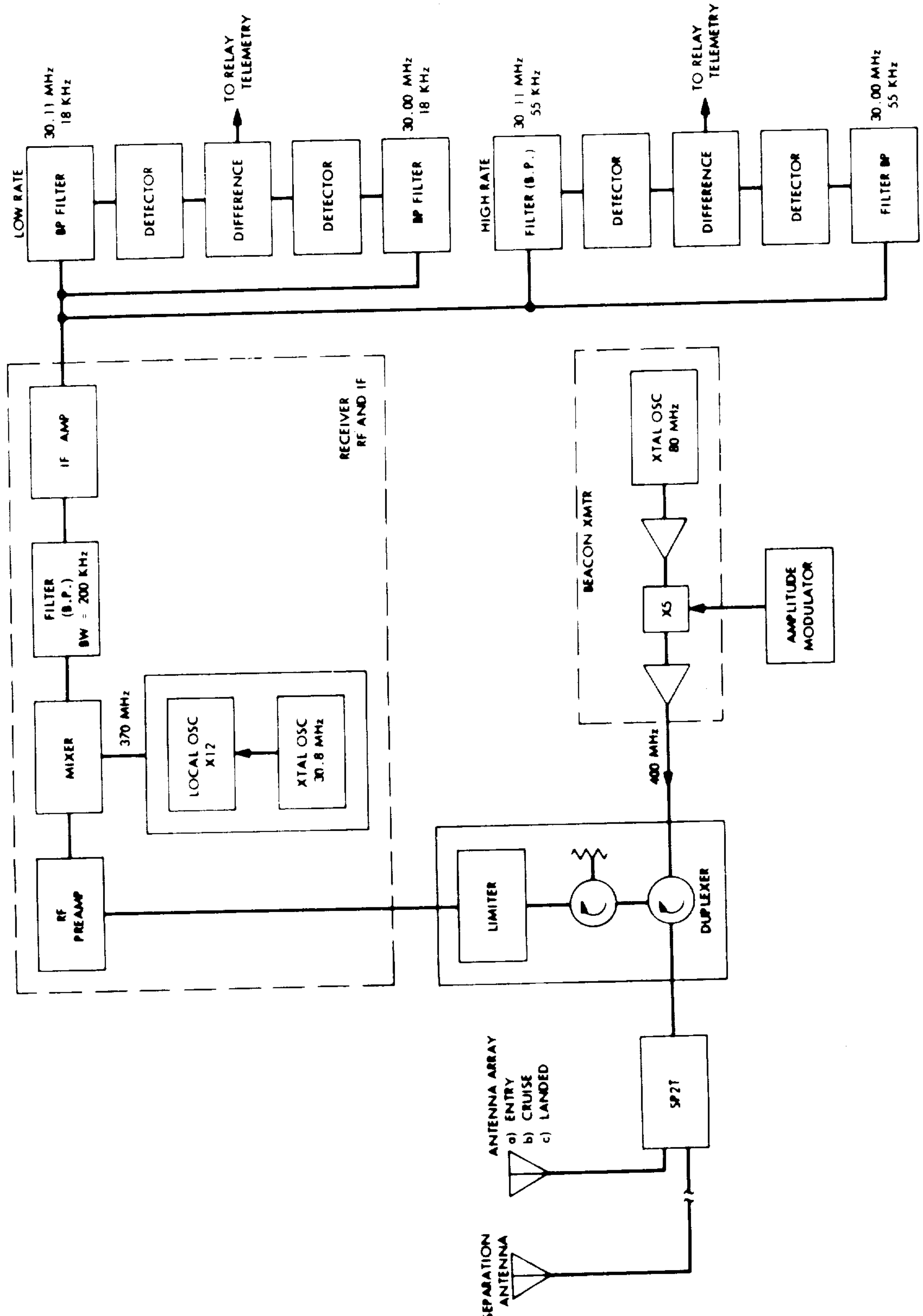

Figure 8C-1. Relay Radio Block Diagram

Table 8C-1. Summary of Relay Radio Parameters

Uplink Freq. (from lander)	400 MHz Non-coherent FSK
Downlink Freq. (from beacon)	400 MHz A. M.
RF Output (beacon)	4 watts (approx.)
Antennas	
Two Low gain Right Hand Circular Polarized	

Weight and Power Requirements*

	Wt (lb)	Power (w)
Antenna Array	9.0	-
SP2T Switch	1.0	1.0
Receiver	3.0	2.0
Transmitter	2.0	10
Duplexer	1.0	
Cables	2.0	-
Totals:	18.0	13

*Note: The above estimates are based on internal R&D tasks.

D. RELAY TELEMETRY SUBSYSTEM

The function of the relay telemetry subsystem (RTS) is to restore and condition lander capsule data and to route it to either the flight telemetry subsystem or the data storage subsystem. Figure 8D-1 shows the relay telemetry subsystem in relation to the FTS and DSS. The RTS consists of two bit synchronizers and the mode control and conditioning circuitry. The bit synchronizers are essentially transition tracking Costas loops capable of acquiring sync in under 100-bit periods. Understandably the presence of sufficient data transitions in the bit stream is required for proper operation. The RTS will handle the various lander capsule data modes in the following manner:

1) The 1050-bph lander capsule status data will be buffered and sub-commutated with orbiter engineering data.
2) The 1348-bps lander capsule check out data prior to lander capsule separation will be hardline and will be routed directly to the FTS where it will be put on a 34.286-KHz subcarrier.
3) The 1348-bps lander capsule post separation and entry data will be restored by the 1348-bps bit synchronizer, then buffered to remove timing jitter and routed to the FTS for real time transmission on the 34.256 KHz subcarrier.
4) The 20 Kbps landed data will be restored by the 20 Kbps bit synchronizer and routed to the DSS. Since the DSS has a readout buffer, timing jitter need not be removed prior to recording.

The mode control circuitry will be controlled by the orbiter CC&S and FCS, however it may be programmed to respond to bit synchronizer lock detector indications when lander capsule data rate changes are expected to take place.

RTS Parameters:

Weight	=	10 lbs
Power	=	5 watts

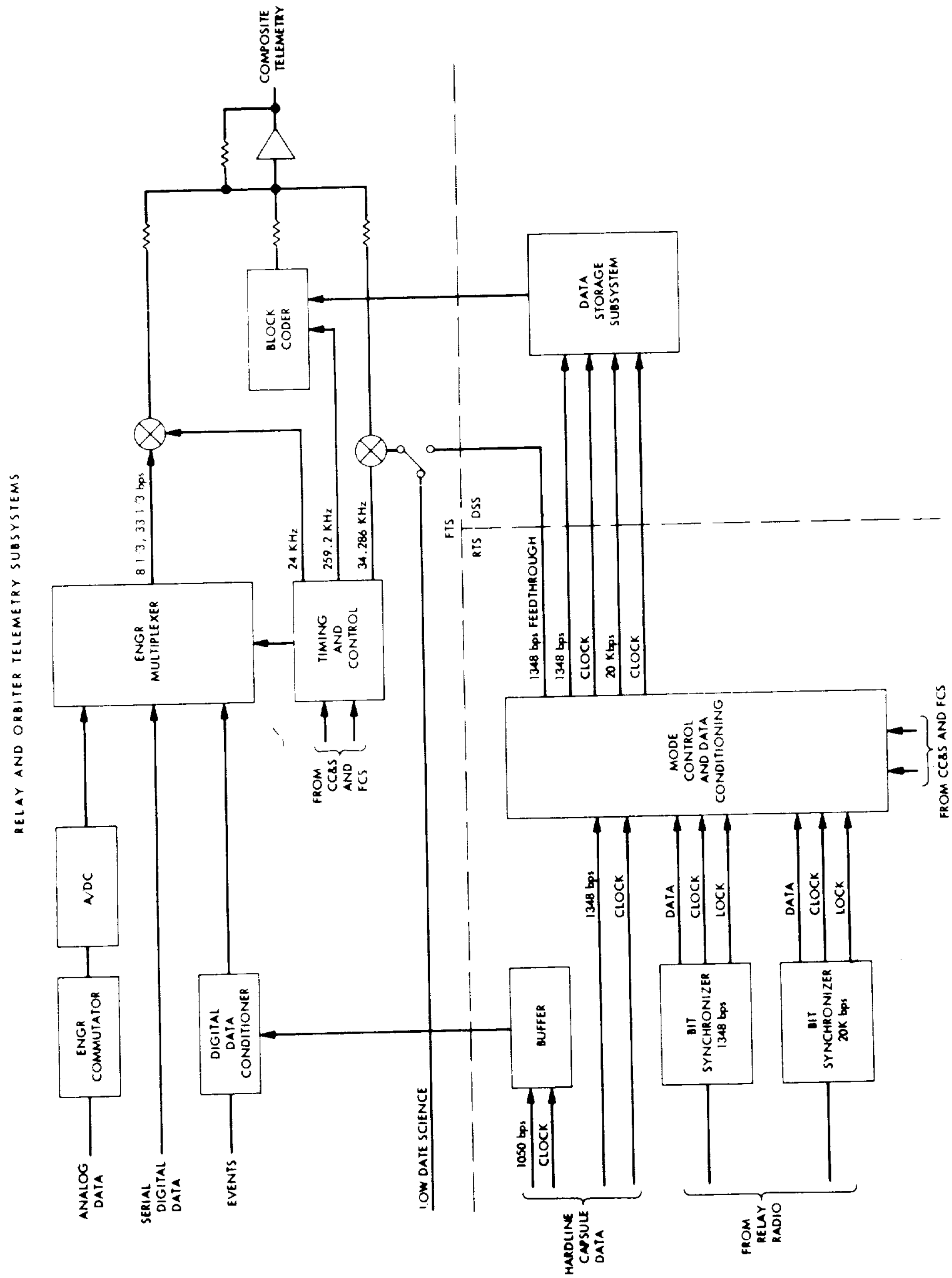

Figure 8D-1. Relay and Orbiter Telemetry Block Diagram

E. FLIGHT COMMAND SUBSYSTEM

The flight command subsystem (FCS) provides the following general functions:

1) Receives from the spacecraft radio subsystem the composite command signal containing the command word information and synchronization information.
2) Acquires phase coherence (lockup) with the synchronization information which is a pseudo-random signal, and establishes a phase reference signal and bit synchronization signal.
3) Demodulates the command word information which is a biphase-modulated sinusoidal subcarrier, using the phase reference established by the synchronization process, and detects and reconstructs the command word data bits.
4) Provides a detector lock signal indicating whether or not detector synchronization with the received signal has been established.
5) Decodes the direct command word address bits and provides a momentary switch closure to the proper spacecraft user subsystem.
6) Decodes the coded command word address bits and directs a serial binary 18-bit word, representing coded command data and bit synchronization information, to the spacecraft central computer and sequencer subsystem.
7) Decodes the quantitative command word address bits and directs a serial, variable length pulse train to the scan subsystem.

A block diagram of the FCS is shown in Figure 8E-1. Three types of commands are transmitted to the spacecraft: (1) direct command (DC) words provide a momentary switch closure to spacecraft subsystems; (2) coded command (CC) words provide serial bits to the spacecraft central computer and sequencer (CC&S) subsystem representing trajectory correction maneuver and update program information; and (3) quantitative command (QC) words provide a variable number of momentary switch closures to the scan subsystem.

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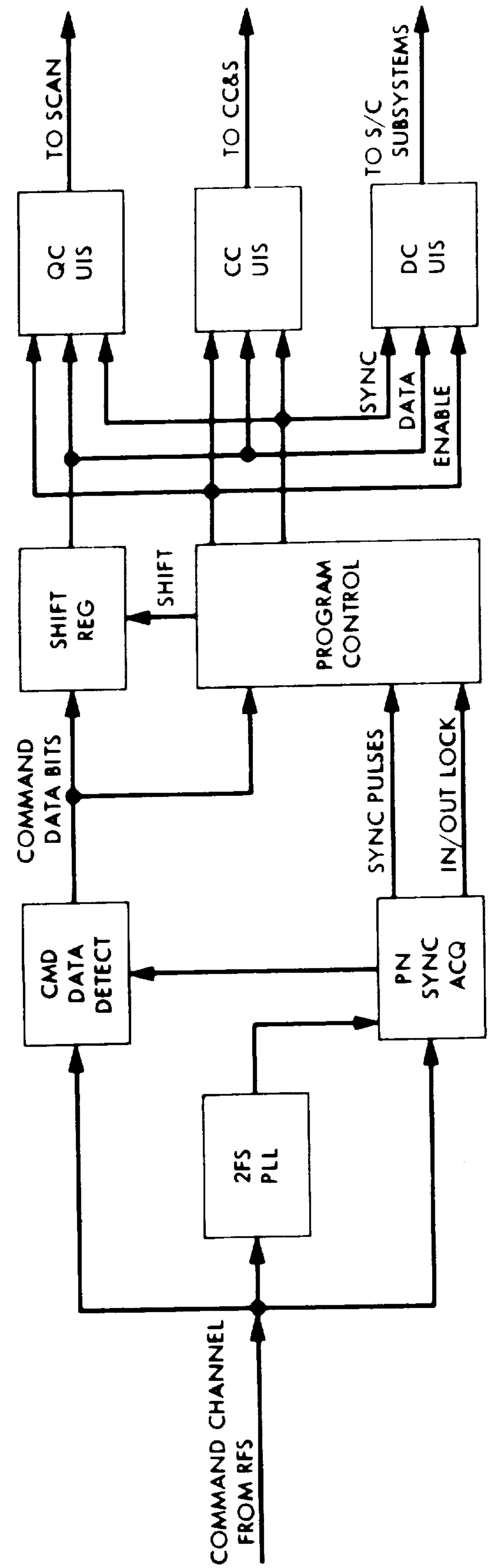

Figure 8E-1. Flight Command Block Diagram

The command detector performs the first four FCS functions listed above. The detector data channel selects the command word information from the composite command signal by means of a bandpass filter at the data subcarrier frequency. The subcarrier is phase-coherently demodulated and filtered. The output is then reconstructed into binary bits by a decision circuit and storage circuit and the resulting command word bits are directed to the decoder. Pseudonoise (PN) command sync information is recovered from the sync subcarrier by the detector sync channel and used to maintain detector lock.

The command decoder processes detected command words, reads the addresses, and outputs either a switch closure to the addressed subsystems in the case of a direct command, a serial bit word representing a coded command (with accompanying sync information) to the CC&S, or a serial pulse train of variable length to the scan subsystem in the case of a quantitative command.

The orbiter FCS will provide several direct command outputs to the capsule as a backup to CC&S. Coded information intended for the capsule will be routed through the CC&S.

FCS Parameters:

Weight	= 11.7 lbs
Power	= 3.5 watts
Command Bit Rate	= 1 bps
Word Length	= 26 bits, 18 bit available to CC command user
Required ST/N_o	= 16.5 DB for $P_{be} \leq 1 \times 10^{-5}$
Lock-up Time	$\leq$ 12.5 minutes

As an example of the capabilities of a flight command subsystem a partial listing of the commands to be utilized on Mariner '71 are given below in Table 8E-1. This command list represents generally what must be anticipated for a Mariner-class orbiter; of course, the flight command subsystem for the Viking orbiter must also include commands which are imposed because of the addition of a lander capsule.

Table 8E-1. Command List

DIRECT COMMANDS

Symbol	Name	Destination	Comment
DC-1	Engineering Mode	FTS	Switches FTS to Engineering data mode
		DSS	SPARE
		PYRO	SPARE
		PWR	SPARE
DC-2	CC&S Readout Select	FTS	Toggles engineering channel from engineering from engineering telemetry to CC&S memory data and vice versa.
DC-3	Playback Mode	FTS	Switches FTS to Playback data mode.
		DSS	Initiates playback of data stored on tape.
		PWR	To be defined
DC-4	DSS Ready Mode	DSS	Stops the tape if it is in motion by switching the DSS to its standby mode.
DC-5	Data Rate Switch	FTS	Toggles engineering channel data rate between 8 1/3 and 33 1/3 bps.
DC-6	FTS Redundant Elements Switch	FTS	Selects alternate set of FTS block redundant elements. Is a toggle command.
DC-7	Power Amplifier Switch	RFS	Selects redundant TWT and power supply. Is a toggle command.
DC-8	Exciter Switch	RFS	Selects redundant exciter unit. Is a toggle command.
DC-9	Ranging On/Off	RFS	Toggles ranging channel On/Off.
DC-10	Transmit/Receive LG#1	RFS	Selects low gain #1 antenna for both reception and transmission.

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Symbol	Name	Destination	Comment
DC-11	Transmit Hi/Receive LG#1	RFS	Selects transmission via high gain antenna and reception via low gain #1.
DC-12	Adaptive Gate Step	A/C	Steps Canopus brightness gate to lower values. Next command following minimum gate setting resets logic to maximum value.
DC-13	Maneuver Inhibit	A/C	Inhibits CC&S commands to A/C and initiates Sun and Canopus reacquisition.
		CC&S	Resets fixed sequencer and switches maneuver logic to the non-tandem standby mode.
		PWR	Turns off power to Gimbal regulator
		PYRO	Safe capacitor banks "C" and "D".
DC-14	Maneuver Enable	A/C	Enables CC&S maneuver commands to A/C. Turns A/C off if previously turned on by DC-13.
		PYRO	SPARE
DC-15	Canopus Gate Override	A/C	Removes brightness gate requirements from Canopus Sensor acquisition logic. This prevents gyros from burning on if roll reference is lost.
DC-16	Record Video Data	SDS	To be determined.
		DSS	Initiates a DSS record sequence.
DC-17	Canopus Cone Angle Step	A/C	Advances cone angle setting to next position.
DC-18	Inertial Roll Control/ Step	A/C	Turns roll gyros on and places roll axis under inertial control. Subsequent DC-18's will generate positive (+) roll increments.

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Symbol	Name	Destination	Comments
DC-19	Canopus Roll Control	A/C	Resets DC-15, DC-18, DC-20, DC-40 and DC-63. Initiates roll search if object not acquired when command received.
DC-20	Roll Control Inhibit	A/C	Removes roll control completely.
DC-21	Roll Search/Step	A/C	Turns roll gyro on and initiates a roll search. If DC-18 is in effect, then all subsequent DC-21's will generate negative (-) roll increments.
DC-22	CW Rotation/Advance Track	DSS	Record or playback: Reverses tape direction and advances DSS to next tape pass if tape direction is CCW. No effect if tape direction is CW. Swithces DSS to ready mode if sldwing.
DC-23	CCW Rotation/Advance Track	DSS	Record or playback: Reverses tape direction and advances DSS to next tape pass if tape direction is CW. No effect if tape direction is CCW. Switches DSS to ready mode if slewing.
DC-24	Unlatch Low-Gain Antenna	PYRO	Deploys low-gain antenna #2.
DC-25	RT Science #1 Mode	FTS	Switches FTS to Real Time Science #1 data mode.
		PWR	Turn on cruise science.
		SCAN	SPARE
		DSS	SPARE
DC-26	Hi-Gain Update Enable	PYRO	Arms capacitor banks "C" and "D".
		SDS	SPARE
		PWR	

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Symbol	Name	Destination	Comments
DC-27	Initiate Maneuver Sequence	CC&S	Initiates tandem or fixed sequencer maneuver depending upon the CC&S maneuver control logic logic state.
		PYRO	SPARE
DC-28	Fixed Scan Position	SCAN	Slew Scan to cone/clock position defined by fixed reference pots. This position is determined prior to launch
		DSS	SPARE
DC-29	Accelerometer Scale Factor	CC&S	Determine whether or not accelerometer output information is divided by 32 in the CC&S. Is a toggle command.
		PYRO	
DC-30	Computer Inhibit	CC&S	Inhibits computer output actuators, halts computer operation and resets all flags.
DC-31	Computer Enable	CC&S	Enables computer output actuators, starts computer operated previously stopped by DC-30 and resets all flags.
DC-32	Computer Maneuver Initiate	CC&S	Initiates a computer-only maneuver sequence if CC&S logic in non-tandem standby mode.
DC-33	Tandem Maneuver	CC&S	Switches the CC&S logic to the tandem standby mode and enables the tolerance detector.
DC-34	Scan Power On/Off	PWR	Toggles power to scan subsystem on/off.
DC-35	Variable Scan Mode	SCAN	Switches scan to variable clock and cone reference pots.
DC-36	SPARE	SDS	
DC-37	Boost Mode Enable/ Inhibit	PWR	Toggles share booster state.
DC-38	Battery Charger On/Off	PWR	Toggles battery charger On/Off.

Symbol	Name	Destination	Comments
DC-39	Tape Slew Mode	DSS	Initiates tape slew mode if DSS logic is initially in ready mode.
DC-40	Gyros Off	A/C	Unconditional command for turning gyro power off. Reset by DC-19.
DC-41	Platform Cone Slew	SCAN	Slews platform in cone direction only to fixed position which is selected prior to launch.
DC-42	TWT High Power	RFS	Switch TWT to its high power output mode.
DC-43	TWT Low Power	RFS	Switch TWT to its ;ow power output mode.
DC-44	RT Science #2 Mode	FTS	Switches FTS to the real-time science #2 data mode.
		DSS	SPARE
DC-45	Platform Unlatch	PYRO	Permanently releases scan platform latching mechanization.
DC-46	TV Cover Deploy	TV	Permanently removes cover from Camera A.
DC-47	DSS Power On/Off	PWR	Toggles 2.4 kHz power to tape On/Off.
DC-48	Stow IRR Mirror	IRR	Sets up mirror sequence logic such that mirror stops at next internal calibrate position. DC-48 condition removed by turning IRR power Off/On.
		TV	SPARE
DC-49	Hi-Gain Antenna Update	PYRO	Actuates pinpuller causing hi-gain to move to its second position.
DC-50	Battery #1 Test Load	PWR	Toggles battery #1 test load on-line/off-line.
DC-51	Transmit/Receive LG#2	RFS	Selects low-gain #2 antenna for both transmission and reception.

a-4

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Symbol	Name	Destination	Comments
DC-52	Transmit/Receive Hi-Gain	RFS	Selects high-gain antenna for both transmission and reception.
DC-53	Downlink Off	RFS	Turns off power to TWT power amplifier.
DC-54	Downlink On	RFS	Turns on power to TWT power amplifier.
DC-55	FTS Maneuver Mode	FTS	Toggles engineering channel between standard data format and maneuver format.
DC-56	Playback Rate #1	DSS	Selects 16.2 Kbps playback rate.
DC-57	Playback Rate #2	DSS	Selects 8.1 Kbps playback rate
DC-58	Playback Rate #3	DSS	Selects 4.05 Kbps playback rate.
DC-59	Playback Rate #4	DSS	Selects 2.025 Kbps playback rate.
DC-60	Playback Rate #5	DSS	Selects 1.0125 Kbps playback rate.
DC-61	Simulate Sun Gate	A/C	Causes A/C logic to issue sun gate signal.
DC-62	Disable Pre-Aim Logic	A/C	Disables autopilot pre-aim circuitry.
DC-63	Roll Gyro On	A/C	Turns on roll gyro only.
DC-64	A/C Maneuver Telemetry	A/C	Toggle function for selecting gimbal or sun sensor telemetry.
DC-65	Pressurize Propulsion "A"	PYRO	Opens nitrogen, oxygen and fuel lines for first time.
DC-66	Pressurize Propulsion "B"	PYRO	Opens nitrogen lines second time.
DC-67	Pressurize Propulsion "C"	PYRO	Opens nitrogen lines third time.
DC-68	De-Pressurize Propulsion "A"	PYRO	Closes nitrogen lines first time.

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Symbol	Name	Destination	Comments
DC-69	De-Pressurize Propulsion "B"	PYRO	Closes nitrogen lines second time
DC-70	Open Propellant Lines "A"	PYRO	Opens oxygen and fuel lines second time.
DC-71	Open Propellant Lines "B"	PYRO	Opens oxygen and fuel lines third time.
DC-72	Close Propellant Lines "A"	PYRO	Closes oxygen and fuel lines first time.
DC-73	Close Propellant Lines "B"	PYRO	Closes oxygen and fuel lines second time.
DC-74	Open Battery Louver	PYRO	
DC-75	Battery #2 Test Load	PWR	Toggles battery #2 test load on line/off line.
DC-76	Select Battery Configuration	PWR	Toggle command for placing battery #2 on line/off line.
DC-77	Charge Battery #1/#2	PWR	Toggle command for selecting which battery can be charged.

QUANTITATIVE COMMANDS

QC-1	Cone Angle + Step	SCAN	Positively steps cone angle ref. value 1-18 increments.
QC-2	Cone Angle - Step	SCAN	Negatively steps cone angle Ref. value 1-18 increments
QC-3	Clock Angle + Step	SCAN	Positively steps clock angle Ref. value 1-18 increments.
QC-4	Clock Angle - Step	SCAN	Negatively steps clock angle Ref. value 1-18 increments.

CODED COMMANDS

Symbol	Name	Destination	Comments
CC-1	Computer Load	CC&S	Program computer. First of pair of CC's required for prog. computer.
CC-2	Computer Load	CC&S	Program computer. Second of pair of CC's required for prog. computer.
CC-3	Word Interr.	CC&S	Fetches one word from memory.
CC-4	Seq. Load	CC&S	Enters fixed sequencer maneuver parameters.
CC-5	Tol. Detector Disable	CC&S	Disables Tol. Detector
CC-6		SDS	

F. POWER SUBSYSTEM

The Viking orbiter baseline power subsystem is essentially a major modification of the Mariner '71 power subsystem. The modifications are necessitated by the following guidelines and constraints:

1) An orbit insertion mode with approximately 167 minutes of off-Sun-line operation and a burn time of three times as long as Mariner '71 is the worst case energy demand period.
2) 100 days of Mars orbital operation with partial Sun occultation during approximately 50 days.
3) The requirement for providing 5 to 182 watts of Viking lander capsule power during the transit and early orbital period.
4) A revised science payload.
5) A greater Sun-Mars distance than Mariner '71 results in a 26% decrease in available solar panel power.

The above factors have revised the power subsystem philosophy established on previous Mariner spacecraft. The battery selected for this mission must be capable of providing approximately 70-75 flight cycles after the 6-7 month transit period. In addition, the battery is also used to supplement the solar panel during the Viking lander capsule checkout period. The solar panel has been sized for the mode of operation requiring battery charging towards the end of the orbital cruise period.

1. Power Requirements

A mission power profile depicting the subsystem power requirements as compared to an operational mode of events is shown in Table 8F-1a and 8F-1b. Total raw power required during the period of orbit insertion maneuver to a post-Sun-occultation mode will vary from 330 to 518 watts. Of significance are the battery charging requirements of 125 watts, the TWT subsystem continuous requirement of 96 watts and the approximate 3-hour Viking lander capsule checkout requirement of 182 watts. During the orbit insertion mode, the orbiter is off the Sun line for 167 minutes and the total battery energy consumed is 1195 watt-hours. Approximately 260 watt-hours of this

Table 8F-1a. Preliminary Power Profile (Baseline)

TABLE 8F
VIKING ORBITER
PRELIMINARY POWER PROFILE
OPERATIONAL MODE (SEE TABLE 8F-1b)

COMPONENT	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17
MAIN INVERTER (2.4 KHz)																	
ENGINEERING																	
FLT TELEM SYST	15.0																
FLT COMMAND SYST	3.5																
CC&S	41.0	18.5															
PYRO	0	1.0															
POWER DISTRIBUTION	2.25																
ATTITUDE CONT No. 1	18.0	30.0	9.0			28.0	9.0			28.0					9.0	28.0	9.0
ATTITUDE CONT No. 2	0					26.0	0				11.0	11.0	26.0	0	0	26.0	0
RADIO FREQ SYSTEM	27.0																
GYRO No. 1	10.0	10.0	0			10.0	0	0	0	10.0					0	10.0	0
SCNE	4.7								23.0	4.7							
THERMAL CONT No. 1	37.0	24.5							0	24.5							
DATA STORAGE SYSTEM	0	10.0							22.0	10.0							
ANTENNA POINTING	3.0																
SCIENCE																	
IR RADIOMETER	0								5.0	0							
IR SPECTROMETER	0								14.0	0							
IR MAPPER	0								14.0	0							
TV	0								37.5	0							
SDS	0	10.0							22.0	10.0							
SUB TOTAL	161.4	159.5	128.5	128.5	128.5	183.4	128.5	128.5	213.7	157.5	168.4	168.4	183.4	157.5	128.5	183.4	128.5
EFFICIENCY	0.903	0.903	0.88	0.88	0.88	0.910	0.88	0.88	0.918	0.903	0.906	0.906	0.910	0.903	0.88	0.910	0.88
TOTAL	178.8	176.6	146.0	146.0	146.0	201.5	146.0	146.0	232.8	174.4	185.9	185.9	201.5	174.4	146.0	201.5	146.0
GYRO No. 2 (400 Hz, 3Ø)	9.0	9.0	0			9.0	0			9.0					0	9.0	0
EFFICIENCY	0.775	0.775	0			0.775	0			0.775					0	0.775	0
TOTAL	11.6	11.6	0			11.6	0			11.6					0	11.6	0
SCNM (400 Hz, 1Ø)	0								9.4	0							
EFFICIENCY	0								0.797	0							
TOTAL	0								11.79	0							
REGULATED CONV (28 VDC)																	
VALVE	0					55 ①	0						55.0	0	0	55 ③	0
GIMBAL	0					25.0	0						25.0	0	0	25.0	0
EFFICIENCY	0					0.870	0						0.870	0	0	0.870	0
TOTAL	0					91.92	0						91.92	0	0	91.92	0
TOTAL REQ'D BOOST REG O.P.	190.4	188.2	146.0	146.0	146.0	305.0	146.0	146.0	244.6	186.0	197.5	197.5	305.0	186.0	146.0	305.0	146.0
EFFICIENCY	0.842	0.842	0.868	0.868	0.868	0.842	0.868	0.868	0.887	0.885	0.887	0.842	0.842	0.842	0.875	0.842	0.875
TOTAL BOOST REG INPUT	226.1	223.5	168.2	168.2	168.2	362.2	168.2	168.2	275.7	210.2	222.7	234.6	362.2	220.9	168.9	362.2	168.9
UNREG POWER REQ'D																	
BOOST REG INPUT	226.1	223.5	168.2	168.2	168.2	362.2 ①	168.2	168.2	275.7	210.2	222.7	234.6	362.7	220.9	168.9	362.2	168.9
THERMAL CONT No. 3	4.1	4.1	20			4.1	20.0						4.1	20.0	20.0	4.1	20.0
BATTERY CHARGING	0.5	0.5	125.0	0.5			125.0	0.5							125.0	0.5	125.0
BOOST REG FAIL. SENSOR	1.5																
TWT SYSTEM	59.0							96.0									
THERMAL CONTROL No. 2	9.3	23.0															
RELAY R.F.	0																
LANDER	15.8	15.8	25.2	15.2	82.7	5.0	25.2	15.2									
SUBTOTAL	316.3	327.4	421.9	287.4	354.9	455.3	421.9	324.4	431.9	366.4	378.9	390.8	502.5	377.1	449.6	502.5	449.6
POWER SOURCE AND LOGIC EFF	0.97																
TOTAL RAW POWER DEMAND (watts)	326.0	338.0	434.0	296.0	366.0	469 ① 357 ②	434.0	334.0	445.0	378.0	391.0	403.0	518.0	389.0	463.0	518 ③ 406 ④	463.0

COMPONENT	18	19	20	21	22	23	24	25	26	27	28	29	30	31	32	33
MAIN INVERTER (2.4 KHz)																
ENGINEERING																
FLT TELEM SYST																
FLT COMMAND SYST																
CC&S																
PYRO																
POWER DISTRIBUTION																
ATTITUDE CONT No. 1	28.0	9.0			28.0						9.0				18.0	9.0
ATTITUDE CONT No. 2																
RADIO FREQ SYSTEM																
GYRO No. 1	10.0	0	0	0	10.0						0				10.0	0
SCNE	23.0	4.7											23.0	4.7	4.7	4.7
THERMAL CONT No. 1	0	24.5											0	24.5	24.5	24.5
DATA STORAGE SYSTEM	22.0	10.0											22.0	22.0	10.0	10.0
ANTENNA POINTING																
SCIENCE																
IR RADIOMETER	5.0	0											5.0	0		
IR SPECTROMETER	14.0	0											14.0	0		
IR MAPPER	14.0	0											14.0	0		
TV	37.5	0											37.5	0		
SDS	22.0	10.0											22.0	10.0		
SUB TOTAL	245.8	128.5	128.5	128.5	157.5	157.5	157.5	157.5	157.5	157.5	128.5	128.5	213.7	140.5	147.5	128.5
EFFICIENCY	0.925	0.88	0.88	0.88	0.903	0.903	0.903	0.903	0.903	0.903	0.88	0.88	0.918	0.89	0.90	0.88
TOTAL	265.7	146.0	146.0	146.0	174.4	174.4	174.4	174.4	174.4	174.4	146.0	146.0	232.8	157.9	163.8	146.0
GYRO No. 2 (400 Hz, 3Ø)	9.0	0	0	0	9.0						0				9.0	0
EFFICIENCY	0.775	0	0	0	0.775						0				0.775	0
TOTAL	11.6	0	0	0	11.6						0				11.6	0
SCNM (400 Hz, 1Ø)	9.4	0											9.4	0		
EFFICIENCY	0.797	0											0.797	0		
TOTAL	11.79	0											11.79	0		
REGULATED CONV (28 VDC)																
VALVE																
GIMBAL																
EFFICIENCY																
TOTAL																
TOTAL REQ'D BOOST REG O.P.	289.0	146.0	146.0	146.0	186.0	186.0	186.0	186.0	186.0	186.0	146.0	146.0	244.6	157.9	175.4	146.0
EFFICIENCY	0.842	0.875	0.875	0.875	0.885	0.885	0.842	0.842	0.842	0.842	0.875	0.875	0.887	0.880	0.842	0.875
TOTAL BOOST REG INPUT	343.3	168.9	168.9	168.9	210.2	210.2	220.9	220.9	220.9	220.9	168.9	168.9	275.7	179.4	208.3	168.9
UNREG POWER REQ'D																
BOOST REG INPUT	343.3	168.9	168.9	168.9	210.7	210.2	220.9	220.9	220.9	220.9	168.9	168.9	275.7	179.4	208.3	168.9
THERMAL CONT No. 3																
BATTERY CHARGING	0.5	125.0	0.5	125.0	0.05						125.0	0.5				125.0
BOOST REG FAIL. SENSOR																
TWT SYSTEM																
THERMAL CONTROL No. 2									13.0				0			13.0
RELAY R.F.			182.0	27.5				0								
LANDER																
SUBTOTAL	499.5	449.6	491.9	370.4	378.7	378.7	389.4	361.9	374.9	374.9	447.4	322.9	416.7	320.4	349.3	447.4
POWER SOURCE AND LOGIC EFF																
TOTAL RAW POWER DEMAND (watts)	515.0	463.0	507.0	473.0	390.0	390.0	401.0	373.0	387.0	387.0	461.0	333.0	430.0	330.0	360.0	461.0

① 15 sec ON; ② 1 hr ON, ③ 1 min ON; ④ 59 min ON

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Table 8F-1b. Preliminary Power Profile (Baseline)

OPERATIONAL MODE	POWER SOURCE	TIME OF OPERATIONAL MODE
1. LAUNCH	BATTERY	L + 81 min
2. SUN ACQUISITION	BATTERY	
3. CRUISE I BATTERY CHG ON (GYROS OFF)	SOLAR PANEL	
4. CRUISE II BATTERY CHG OFF	SOLAR PANEL	
5. CRUISE III V.L.C. CHECKOUT	SOLAR PANEL	
6. MANEUVER	BATTERY	2 hr NOMINAL
7. CRUISE I BATTERY CHG ON	SOLAR PANEL	
8. CRUISE II BATTERY CHG OFF	SOLAR PANEL	
9. ENCOUNTER	SOLAR PANEL	P_1 (20 Mar '74)
10. GYRO WARM UP ON	SOLAR PANEL	P_1 - 3 hr 11 min
11. ORBIT INSERTION MANEUVER	SOLAR PANEL	P_1 - 2 h 52 m to P_1 - 1 h 55 m
12. ORBIT INSERTION MANEUVER	BATTERY	P_1 - 1 h 55 m to P_1 - 23 m
13. ORBIT INSERTION MANEUVER BURN	BATTERY	P_1 - 23 m to P_1 + 23 m
14. ORBIT INSERTION MANEUVER AND UNWIND	BATTERY	P_1 + 23 m to P_1 + 51 m
15. ORBIT CRUISE WITH V.L.C.	SOLAR PANEL	P_1 + 51 m to P_2 - 30 m
16. ORBIT TRIM WITH V.L.C.	BATTERY	P_2 - 30 m to P_2 + 30 m
17. ORBIT CRUISE WITH V.L.C.	SOLAR PANEL	P_2 + 30 m to P_3 - 45 m
18. ORBIT CRUISE WITH V.L.C. AND SCIENCE	BATTERY	P_3 - 45 m to P_3 + 33 m
19. ORBIT CRUISE WITH CAPSULE	SOLAR PANEL	P_3 + 33 m to P_4
20. V.L.C. PRESEPARATION CHECKOUT	SP/BATTERY	P_8 to P_8 + 3 h
21. V.L.C. EVALUATION AND ASSESSMENT	SOLAR PANEL	P_8 + 3 h to P_8 + 12 h
22. GYRO WARM UP (BC JET MAN.)	SOLAR PANEL	P_{10} - 5 h 6 m
23. BC JET MANEUVER	SOLAR PANEL	P - 3 h 58 m to P - 3 h 42 m
24. BC JET MANEUVER	BATTERY	P - 4 h 42 m to P - 3 h 26 m
25. V.L.C. SEPARATION MANEUVER	SOLAR PANEL	P - 3 h 26 m to P - 2 h 58 m
26. V.L.C. SEPARATION	BATTERY	P - 2 h 58 m to P - 2 h 4 m
27. V.L.C. SEPARATION MANEUVER UNWIND	SOLAR PANEL	P - 2 h 4 m to P - 1 h 52 m
28. ADAPTER JET MANEUVER	SOLAR PANEL	P - 1 h 44 m to P - 1 h 28 m
29. ENTRY RELAY	SOLAR PANEL	P - 29 m to P + 5 m
30. ORBITAL SCIENCE ON	SOLAR PANEL	P - 60 m to P - 20 m
31. ORBITAL SCIENCE PLAYBACK	SOLAR PANEL	P + 11 m to P + 7 h
32. ORBITAL CRUISE (SUN OCCULTATION)	BATTERY	2 hr MAXIMUM
33. ORBITAL CRUISE WITH BATTERY CHG	SOLAR PANEL	22 hr 42 m

total are due primarily to the additional 30 minutes of orbit insertion burn time occurring with the single motor operation. Figure 8F-1 depicts this period which is the basis for the sizing of the battery.

2. Solar Panel

The primary power source selected for this mission has been based on the utilization of Mariner '71 solar cell technology. However, when projecting this technology to a 20 March 1974 arrival date, a significant loss in performance occurs due primarily to the increased distance of the Sun with respect to Mars. Figure 8F-2 compares the Sun-Mars distance for a Mariner '71 and with that of the Viking arrival date.

At the time of arrival 20 March '74, Mars is approximately 1.62 AU from the Sun, equivalent to 53.2 mw/cm^2. Solar cell performance is estimated at 5.80 w/ft^2 (-10° C). However, a conservative estimate of 10% degradation due to solar flare activity and an 8% reduction due to uncertainties in predicting the nominal power due to measurement limitations reduced the specific power to 4.75 w/ft^2 (-10° C) at arrival. The specific power is further reduced to 4.45 w/ft^2 (-16° C) 90 days after arrival. An additional 2% radiation loss is assumed for this time period. Based on a preliminary load analysis for this period, 461 watts of raw power are required. On this basis, the total active solar panel area required would be approximately 104 ft^2. Total power at the time of arrival is 494 watts and degrading to 464 watts after 90 days. Note: the Mariner '71 panel consisting of 74.7 ft^2 of active area would produce at this point in time, approximately 350 watts. Table 8F-2 summarizes the solar panel performance characteristics at near Earth and at Mars including 90 days after the arrival date.

The present solar panel configuration (Ref. Figure 7B-1) consists of four rectangular panels. Total panel weight, not including structure and based on Mariner '71 technology is 69.68 lbs for 104 ft^2 of active area.

3. Battery

The desire to achieve a total of approximately 70-75 flight discharge/charge cycles including approximately 50 Sun occultation periods has

VIKING ORBITER
PRELIMINARY POWER PROFILE (BASELINE)
(ORBIT INSERTION MANEUVER TO LANDER EVALUATION)

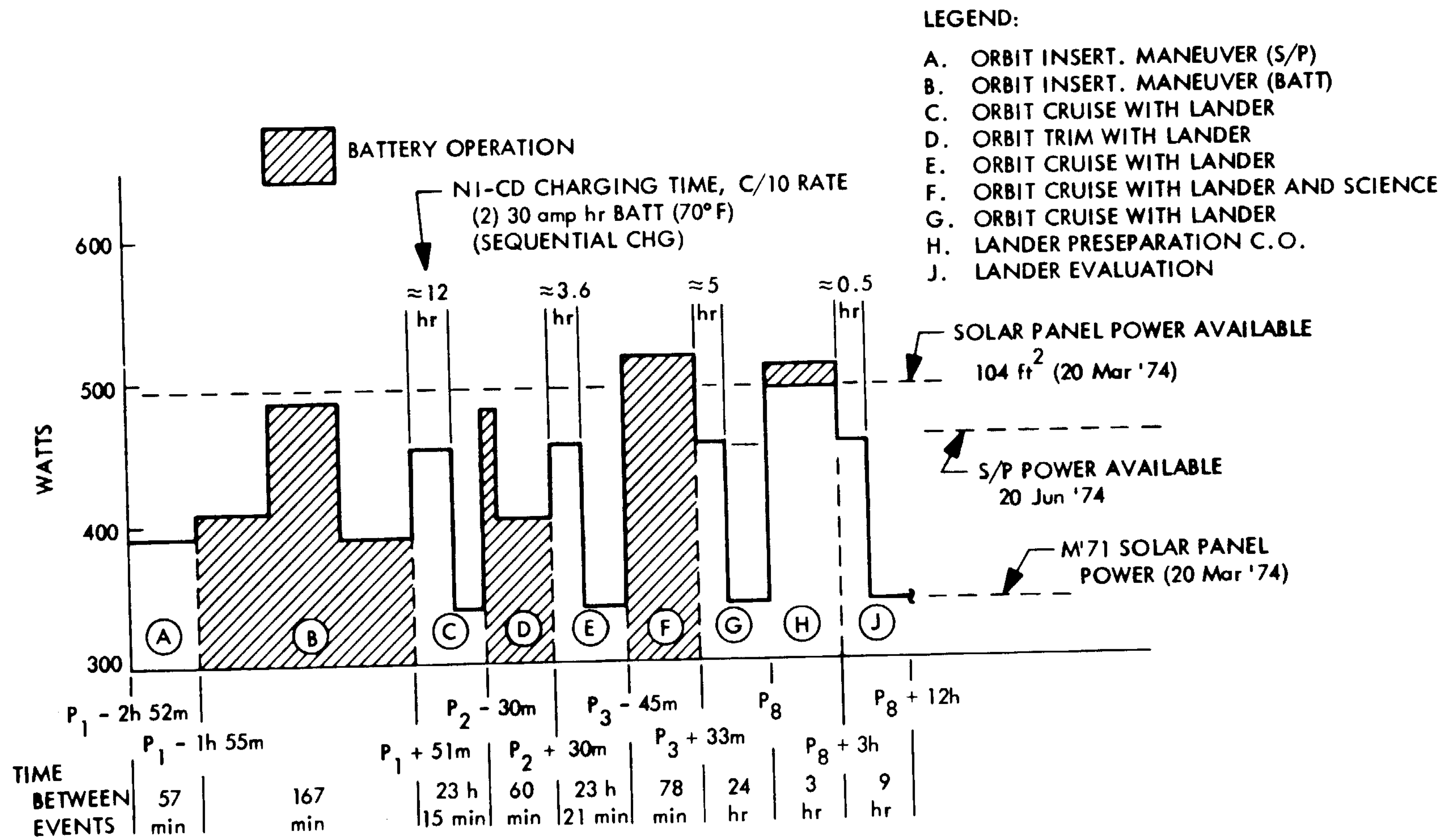


Figure 8F-1. Preliminary Power Profile - Orbit Insertion Maneuver to Viking Lander Capsule Evaluation (Baseline)

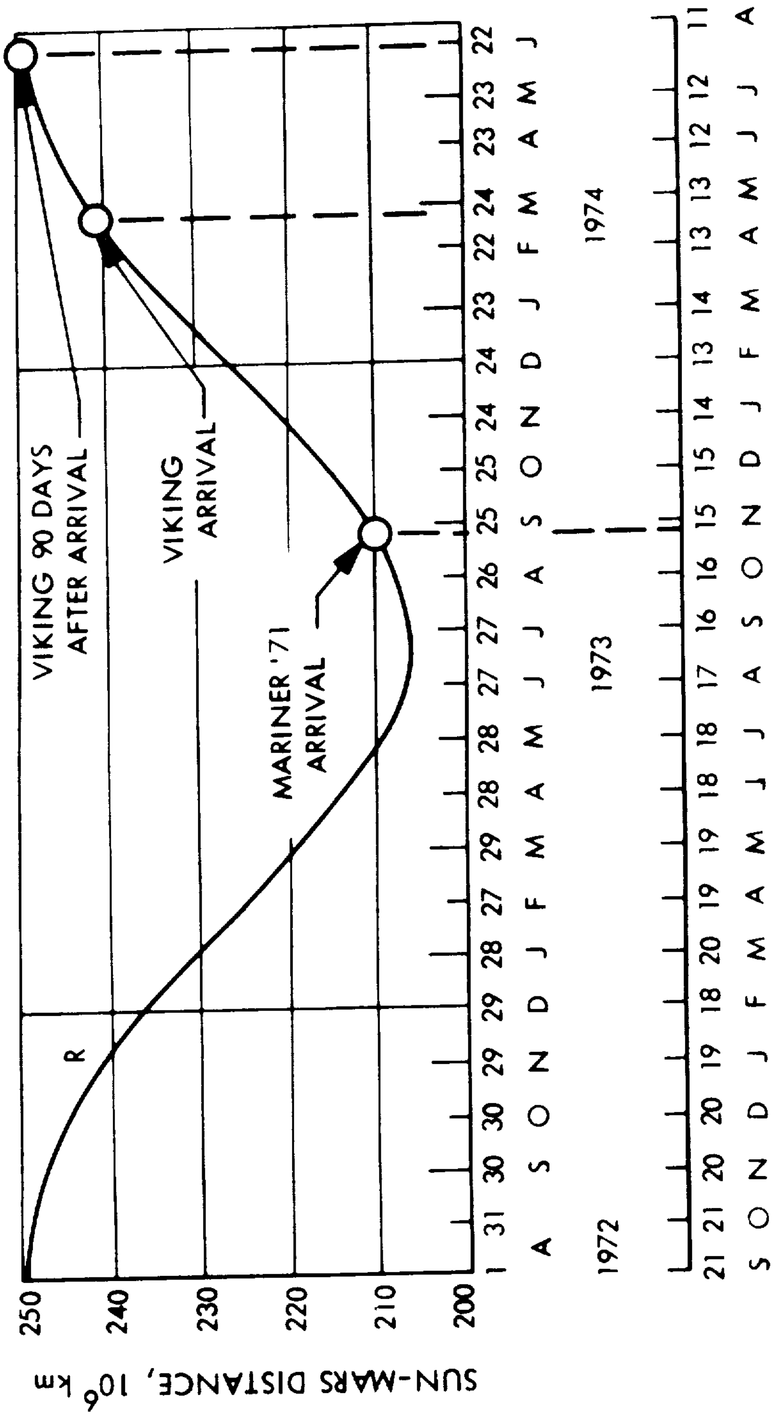


Figure 8F-2. A Comparison of the Sun-Mars Distance for the Viking and Mariner '71 Arrival Dates

Table 8F-2. Preliminary Solar Panel Performance (Baseline)

VIKING ORBITER
PRELIMINARY SOLAR PANEL PERFORMANCE
(BASELINE)

	ARRIVAL DATE	90 DAYS LATER
	20 Mar '74	20 Jun '74
● DISTANCE FROM SUN	1.62 A.U.	1.67 A.U.
● SOLAR INTENSITY	53.2 mw/cm^2	50.4 mw/cm^2
● SPECIFIC POWER		10.4 w/ft^2
NEAR EARTH (60° C) (NO RAD DEGRADATION)	10.4 w/ft^2	10.4 w/ft^2
AT MARS	4.75 w/ft^{2*} (-10° C)	4.45 w/ft^{2**} (-16° C)
● TOTAL POWER AVAILABLE NEAR EARTH	1082 watts	1082 watts
AT MARS	494 watts	464 watts
● TOTAL ACTIVE AREA	104 ft^2	SAME
● SOLAR CELL WEIGHT (NOT INCL STRUCT)	69.68 lb	SAME
● TYPE SOLAR CELL	10.2% (+28° C, 18 mil N/P)	SAME
● COVER GLASS FILTER THICKNESS	20 mil	SAME

*INCLUDES AN ESTIMATE OF 10% DEGRADATION DUE TO SOLAR FLARE ACTIVITY AND AN 8% REDUCTION DUE TO UNCERTAINTIES IN PREDICTING THE NOMINAL POWER DUE TO MEASUREMENT LIMITATIONS

**ASSUMES AN ADDITIONAL 2% RADIATION DEGRADATION necessitated a deviation from the present Mariner '71 battery philosophy. Hermetically sealed nickel cadmium batteries have been selected for the Viking mission because of their cyclic capability. Total energy consumed during a worst case period of operation (167 minutes off the Sun line for the orbit insertion mode of operation) is 1195 watt-hour . Total battery capacity is estimated to be 60 ampere-hour and is based on a 70% depth of discharge. During the maximum 2-hour Sun occultation periods, a total of 720 watt hour are consumed representing a depth of discharge of approximately 42%.

Because of the desire to reduce the solar panel area required for battery charging, two 30-ampere-hour batteries have been selected which are discharged in parallel but charged sequentially. Each 30-ampere-hour battery consists of 24 cells with a battery potential ranging from 32. 5 to 25. 8 volts DC. The weight of each battery including a 15% case weight is estimated to be 69 lbs. Total battery weight is 138 lbs. Total battery volume is estimated to be approximately 1.4 cu. ft.

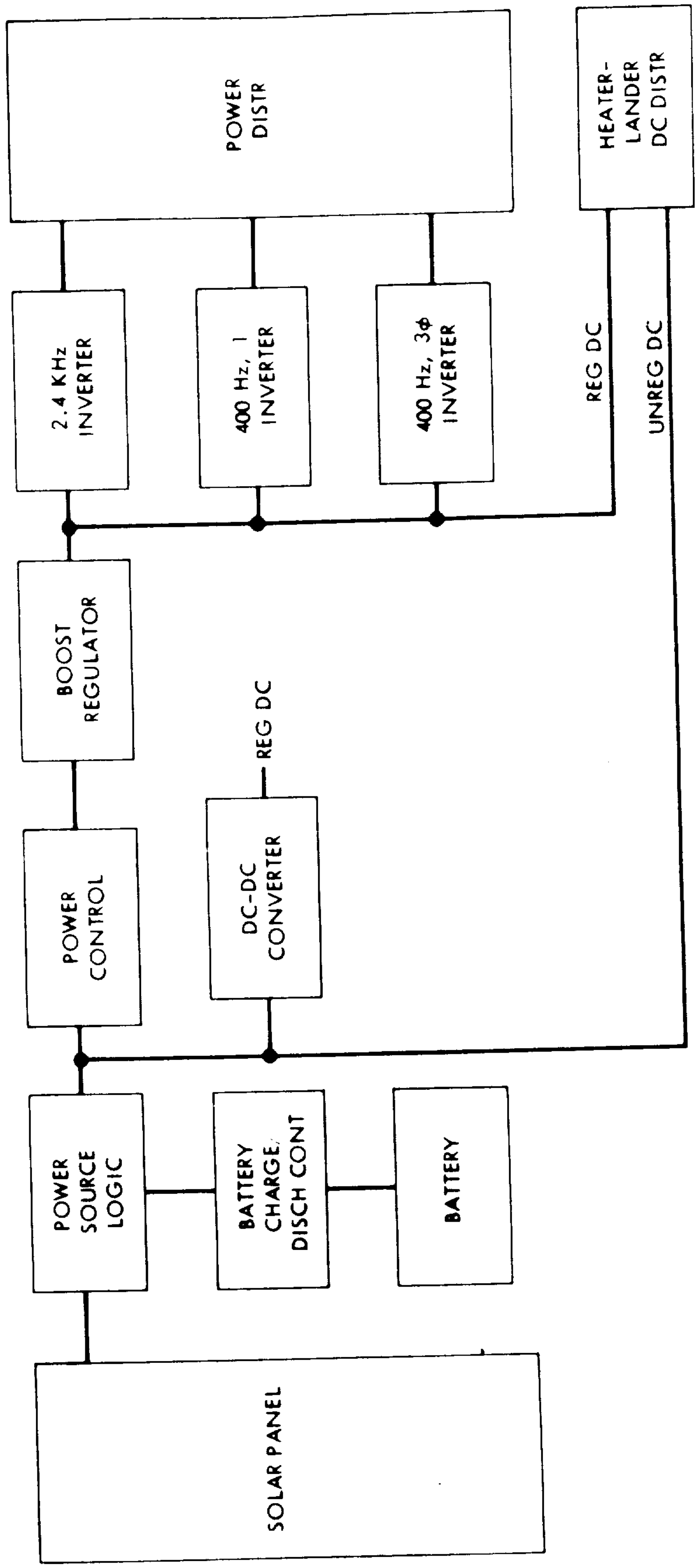
a. Battery Charging

Battery chargers are provided for replacing energy consumed during normal battery operation. In addition, periodic charging is performed during the transit period to replace energy lost due to the charge retention characteristics of nickel cadmium batteries. Trickle charging during this period is also an option.

The charging of the 30-ampere-hour batteries in this configuration is accomplished with constant current at the C/10 rate (3 amp) and constant potential, current limited. In the interests of saving approximately 28 ft^2 of solar panel area, charging of the two batteries is accomplished sequentially. Sufficient time is available throughout the mission to accomplish this. The battery charger/discharge control unit weight is estimated to be 4. 5 lbs each or a total of 9. 0 lbs.

4. Power Conditioning

The orbiter baseline power conditioning shown in Figure 8F-3, Viking Functional Block Diagram, utilizes the Mariner '71 technology with



NOTE: REDUNDANCY NOT SHOWN

Figure 8F-3. Preliminary Power Subsystem Functional Block Diagram (Baseline) some modifications to the booster regulator and main inverter to accommodate increased power requirements. The system provides regulated dc power, derived from a DC-DC converter booster regulator to individual inverters which provide -- (a) 2.4-KHz, single-phase, square-wave power; (b) 400-Hz, three-phase, quasi-square-wave power; and (c) 400 Hz, single-phase, square-wave power. A separate DC-DC converter provides regulated 28-VDC power from the unregulated raw bus.

a. Booster Regulator

The main booster regulator is designed to provide regulated power to all the Viking orbiter loads with the exception of the TWT system, the propulsion valves and gimbals, and the Viking lander capsule. The output voltage is a regulated 56 VDC at a tolerance of ±1%. Due to the increase in the orbiter power requirements, the Mariner '71 main booster regulator must be upgraded from a 250 watt maximum to approximately 305 watts for 45 minutes of operation. Tables 8F-1a and 8F-1b summarize the booster regulator output power requirements. An identical standby booster regulator serves as a backup to the main booster regulator and will provide power to the aforementioned orbiter loads in case of failure of the main booster regulator. Switchover is controlled by onboard detection of over -- or under -- voltage at the output of the main booster regulator. The weight of each modified booster regulator is estimated to be 6.0 lbs. Total weight is estimated to be 12.0 lbs.

b. Main Inverter

The main inverter output is designed to provide 2.4-KHz ±.01% single-phase, square-wave power at 50 VRMS $^{+2}_{-3}$% from a regulated 56-VDC ±1% input. The Mariner '71 main inverter must be modified, due to the increase in orbiter power requirements, to provide a maximum of 250 watts for 78 minutes of operation from the present 200 watt maximum output. Tables 8F-1a and 8F-1b summarize the main inverter output power requirements. In the event of a failure, the system will switch to a standby inverter with identical output characteristics. Timing signals to the central computer and sequencer are provided from the main inverter. The weight of each modified main inverter is estimated to be 3.0 lbs. The total weight is estimated to be 6.0 lbs.

c. 400-Hz Inverter

The Mariner '71 400-Hz inverter, including both single-phase and three-phase outputs, may be used without modification. The output characteristics of the single-phase portion, including both amplitude and frequency are 28 VRMS ±5% square wave and 400 Hz ±0.01% in the normal mode, respectively. Average power output is 15 watts at 28 V with a peak loading of 21 watts for a maximum of 60 seconds. The 400-Hz three-phase power output characteristics, including both frequency and amplitude, are 400 Hz ±0.01% in the normal mode and 27.2 VRMS ±5% line-to-line, quasi square wave, respectively. Average power of the three phase output is 12 watts with a peak of 15 watts. The weight of the 400-Hz inverter is 3.51 lbs.

d. DC-DC Converter

The Mariner '71 DC-DC converter is utilized for the propulsion valves and gimbals. This unit converts the unregulated bus voltage to a regulated 28 VDC. Power output required is 80 watts. The weight of the DC-DC converter is estimated to be 6.0 lbs.

5. Power Distribution

The Viking orbiter power distribution philosophy is basically that of Mariner '71 with modifications for a battery charge/discharge control and revised Viking orbiter power requirements. The solar panel output is distributed to the power source logic unit whereupon the unregulated DC power is distributed to the DC power distribution unit for Viking lander capsule usage. Unregulated or raw DC power is also distributed to the DC-DC converter as well as the boost regulators, battery chargers and the telecommunications system for separate conversion. The output of the two nickel cadmium batteries is distributed through battery discharge control units for equal sharing of the batteries during discharge.

a. Power Source Logic (PSL)

The power source logic unit contains the DC bus from which the solar panel and battery output are connected. The unit also contains the kinetic switch for selection of internal or external power, the isolation diodes for the solar panel sections and a portion of the power subsystem telemetry transducers. The PSL unit weighs 8.0 lbs.

b. Power Control

The power control subassembly contains the failure sense circuit to detect out-of-tolerance conditions in the main power chain, the power control relay to switch from the main power chain to the redundant or standby power chain, the attitude control relay that switches power to the attitude control system and additional power subsystem transducers. The weight of this unit is 2.35 lbs.

c. Heater -- Viking Lander Capsule DC Power Distribution

This modified Mariner '71 unit contains the fuses for the power to the orbiter heaters, a current telemetry transducer and relays controlled by the flight command system for the battery test load. Unregulated power to the lander is distributed through a relay in this unit controlled by the CC&S. Short circuit protection and the prevention of reverse flow of current from the Viking lander capsule are also contained in this unit. The heater -- Viking lander capsule DC power distribution weight is estimated to be 2.0 lbs.

d. Power Distribution Subassembly (PDS)

The PDS receives signals from the flight command system and the CC&S for distribution of the orbiter power. The signals are then processed in the module to actuate magnetic latching relays for power distribution. The input control signals are isolated switch closures between the command and the command line common of two types: pulse saturated transistor and pulse closure of relay contacts. The PDS unit weighs 1.83 lbs.

6. Subsystem Weight

A preliminary weight breakdown of the orbiter power subsystem combined with the design status of each component is shown in Table 8F-3. The total weight of this system is approximately 60% heavier than that of the Mariner '71 power subsystem. This is due primarily to the increased solar panel area combined with the increase in battery capacity required.

Table 8F-3. Preliminary Power Subsystem Weight (Baseline)

		DESIGN STATUS
SOLAR PANEL 104 ft^2 (NOT INCL STRUCT)	69.68	NEW DESIGN
BATTERIES - Ni Cd (2) (INCL 15% CASE wt)	138.00*	NEW DESIGN
BATTERY CHARGE/DISCHARGE CONTROL (2)	9.00	NEW DESIGN
MAIN INVERTER (2)	6.00	MODIFIED
BOOSTER REGULATOR (2)	12.00	MODIFIED
HEATER-LANDER DC POWER DISTRIBUTION	2.00	MODIFIED
POWER CONTROL	2.35	MM'71
POWER SOURCE LOGIC	8.00	MM'71
POWER DISTRIBUTION	1.83	MM'71
400 Hz INVERTER (1ϕ, 3ϕ)	3.51	MM'71
DC/DC CONVERTER	6.00	MM'71
	258.37 lbs	

*ASSUMES NO SUN OCCULATION IN SEQUENCE WITH THE ORBIT INSERTION MODE

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G. CENTRAL COMPUTER AND SEQUENCER

1. Introduction

The Central Computer and Sequencer (CC&S) provides event timing and sequencing of all spacecraft functions which must be generated on a time-dependent basis. This sequencing is generated by a special purpose programmable computer with fixed sequencer redundancy in the maneuver mode. Timing and sequencing (excepting the fixed sequencer) is programmed into the CC&S prior to launch and can be modified during flight by coded commands (CC). The Central Computer and Sequencer functional block diagram, shown in Figure 8G-1, consists of a special purpose computer and a fixed sequencer. The estimated CC&S weight is 27. 5 lbs.; it consumes 20 watts of power, (41 watts at launch) and occupies a volume of 600 $in.^3$

2. Baseline Mechanization Description of CC&S.

a. Computer Mechanization

Approximately 80 percent of the total CC&S subsystem is organized into a computer which may be classified as a stored program, serial operating, special purpose digital machine. The design as implemented is an outgrowth of Voyager studies tailored to state-of-the-art techniques available within schedule and resources to support the Mariner Mars 1969 mission. The computer, utilizing a prelaunch stored program which is inflight variable, can generate all required CC&S mission functions for Mariner Mars 1969. With additional program storage capability and event actuators, the design flexibility is such that all functions estimated for Viking in the JPL "baseline design" could be implemented. This design approach has been selected to correct inherent flexibility deficiencies in fixed sequencer designs previously utilized with the objective of developing an overall design requiring minimum change and qualification for future similar missions. The computer, as shown in Figure 8G-2, consists of the following basic elements presented in the order by which they will be defined:

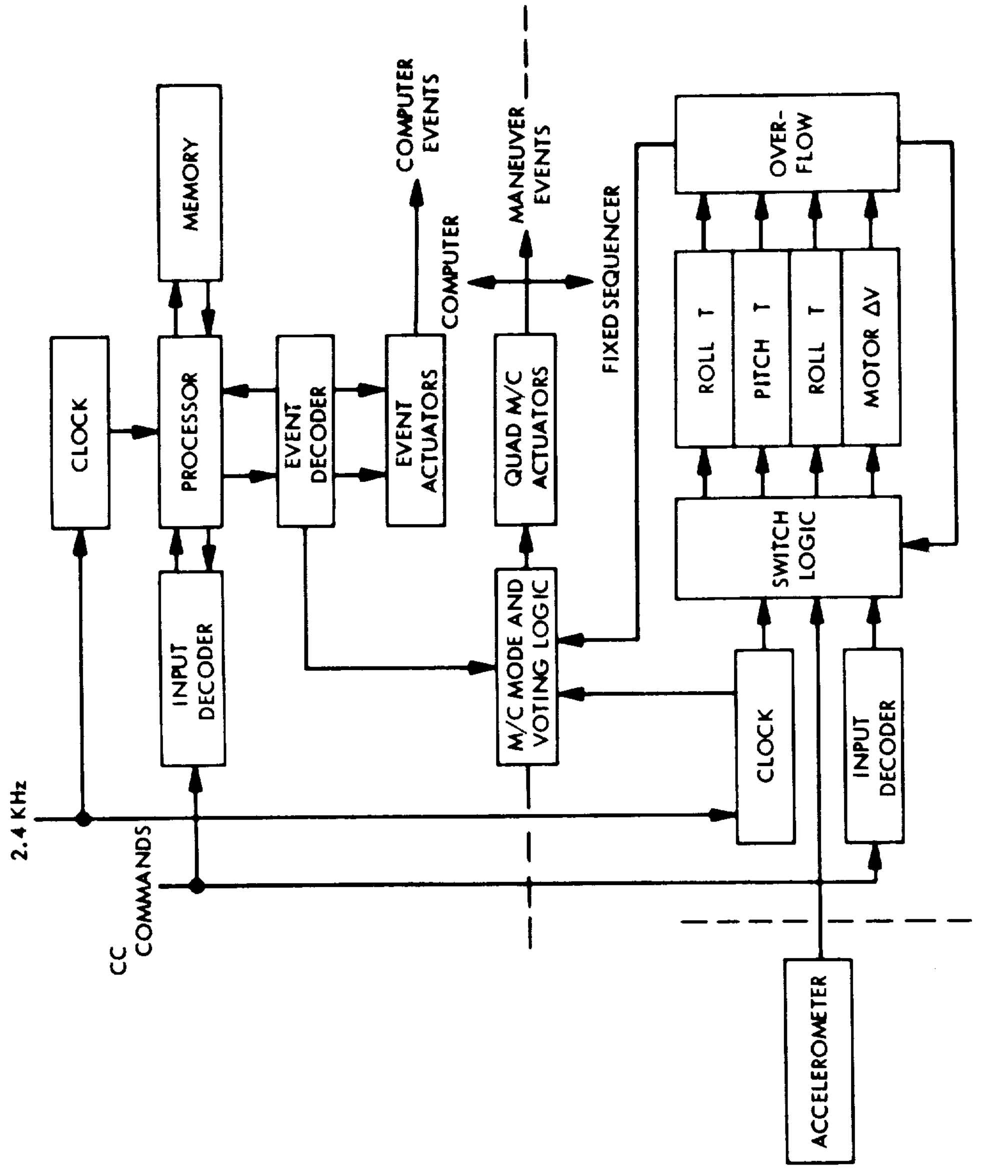

Figure 8G-1. Organization Block Diagram of Computer and Sequencer

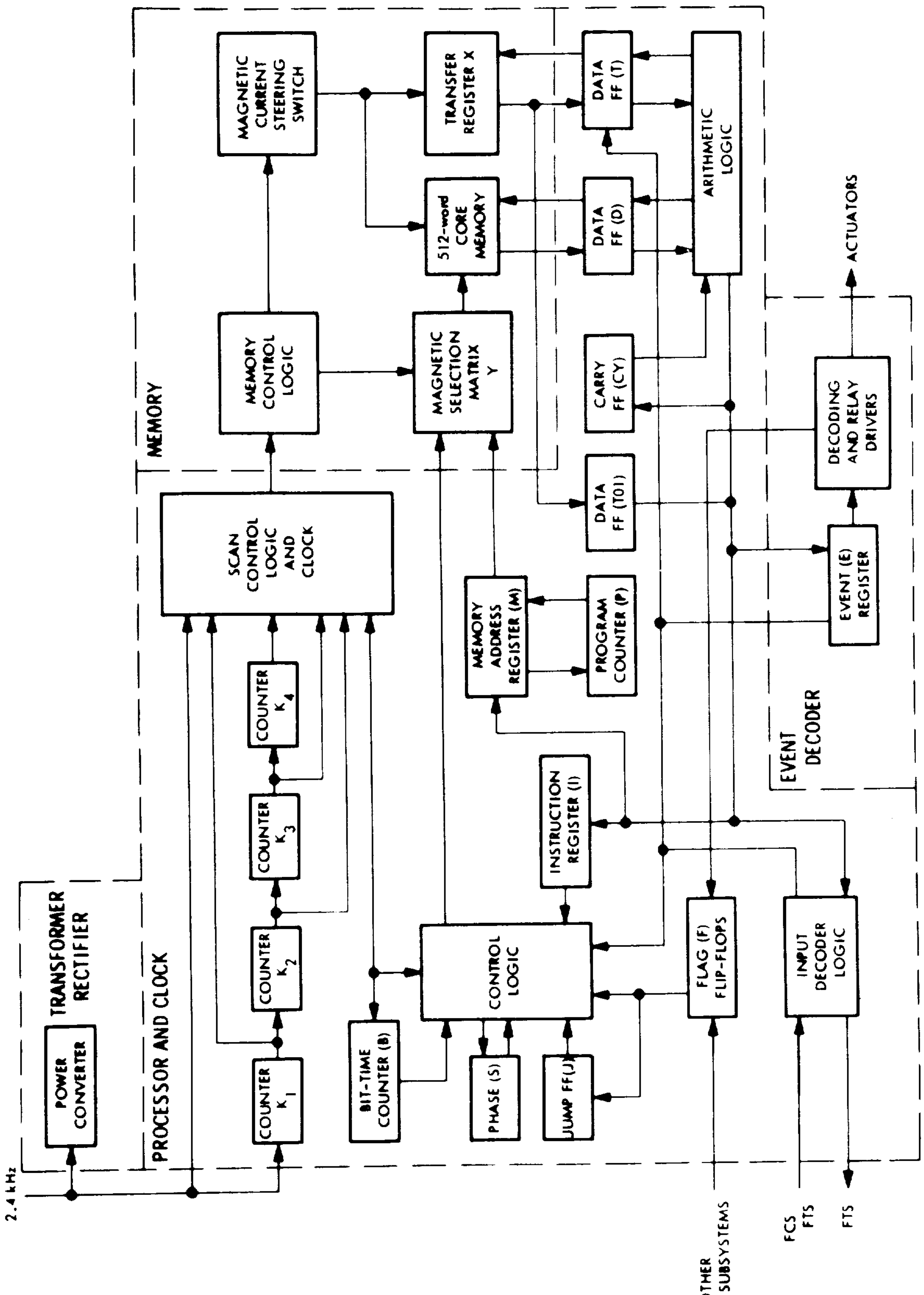

Figure 8G-2. Block Diagram of Computer

1) Input decoder
2) Clock
3) Processor
4) Memory
5) Event decoder
6) Volatility protection
7) Power conversion

(1) Computer input decoder

The computer input decoder has the primary function of receiving inflight coded commands (CC's) from the FCS and routing the contained data into the computer memory. Secondary functions include controlling inflight readout of the computer memory via the FTS and prelaunch memory loading and readout via the CC&S-OSE as well as direct decoding capability for CC commands. The reception and routing of CC words, for memory loading and readouts, is done in a real-time mode in that normal timekeeping routines are temporarily interrupted in the computer. Such command loading is thus restricted to intervals when the CC&S is in cruise mode (one-hour operation cycles) unless time losses or subsequent reprogramming can be tolerated. This loading restriction saves considerable circuitry within the CC&S at essentially no penalty to the spacecraft since all loading cycles would normally occur during intervals when the CC&S is in cruise mode. A timed interlock is provided in the input decoder to return the computer to real-time operation should the CC word be interrupted before completion.

(2) Computer Clock

The computer clock serves as the source of reference signals required by the processor and the memory. It accepts a 2.4-KHz reference frequency and divides it down to the following: 1) provide signals required by the processor scan control, 2) provide the read/write digit pulse rate for the memory, 3) provide the time base for counting down the motor burn duration, and 4) provide a pulse every hour to the A/C subsystem (referred to as the Canopus gate cyclic).

(3) Computer Processor

The processor contains facilities for: addressing, fetching and storing of binary information in the memory, sequencing of instructions stored in memory, arithmetic and time-dependent processing of memory information, and controlling the initiation of relay actuations. The data cycles executed by the processor are, of course, directly related to the information as fetched from the memory. The processor is initiated via the scan control logic (in the control logic block of Figure 8G-2) during each hour pulse or during each minute pulse if the minute toggle has been set (by a DMJ instruction) or during each second pulse if the second toggle has been set (by a DSJ or TAB instruction). The processor also starts during any external interrupts such as power turn on, the loading of memory word from the FCS, etc. Any external interrupts will set the second toggle. The scan control logic is essentially a hardwire executive in the processor.

The internal CC&S instruction word format consists of twenty-two (22) bits, four (4) bits of operation code, two address codes of nine (9) bits each called A and B.

(LSB)		(MSB)
Op Code	Address A	Address B
1 — 4	5 — 13	14 — 22

INSTRUCTION FORMAT

The internal CC&S time/event address word format consists of twenty-two (22) bits, thirteen (13) bits of time magnitude, and nine (9) bits of event address.

(LSB)	(MSB)
Time Magnitude	Event Address
1 — 13	14 — 22

TIME/EVENT ADDRESS FORMAT

The internal CC&S data format is twenty-two (22) bits.

(LSB)		(MSB)
1	DATA FORMAT	22

(4) Computer Memory

The memory provides ferrite core storage and associated input/output circuitry for the computer program. Storage capacity is 512 words of 22 bits each. Words may be randomly addressed, but bits must be sequentially read from each word. Readout of each bit is destructive and requires restoration of the stored state with each bit readout cycle. A special one-word storage register (referred to as the X or transfer register) is included and is addressed and read in parallel with each and every one of the 512 main memory words.

(5) Computer Event Decoder

The event decoder converts event address information furnished by the processor into relay actuations. A partial decoding matrix capable of decoding 72 discrete states from a nine-bit address has been mechanized so that from one to three of these discrete states in certain restricted combinations may be simultaneously selected.

(6) Computer Volatility Protection

To help assure the ability of the computer to function from launch to the end of mission without reliance on ground command and to minimize the possibility of computer interference due to abnormal performance during mission critical functions, it is mandatory to protect program storage to the maximum extent practicable. The following factors were used in the development of the CC&S: minimizing, by program, the intervals during which the CC&S is volatile to excessive power transients (greater than 20 percent) or power dropout; and to sense such transients so that, should they occur when program is volatile, operation will be automatically stopped until the program can be analyzed and corrected by ground command as necessary. Studies of typical mission profiles and CC&S related programs indicate the program is volatile about 0.003 percent of the mission time. This, of course, occurs when the processor is scanning the memory storage and is executing instruction cycles dependent on data contained in semiconductor circuitry. It is a near certainty that excessive transients or power dropout during scan would destroy large quantities of program data which could force the computer into an uncontrolled performance mode. A power monitor (tolerance detector) is provided in the CC&S design which detects all transients in excess of two-microseconds duration and -20-percent amplitude variance from normal input power.

(7) Computer Power Converter

The computer power converter consists of transformers, rectifiers, surge suppression chokes and switching circuitry, and pulse power storage capacitors. All DC power is converted from 2.4-KHz spacecraft power without secondary regulation. Turn-on surge protection within 150-percent overload is accomplished by chokes in the +4-VDC and -4-VDC supplies. Relay resistance stepdown is provided in the +8-VDC and +28-VDC supplies. Pulsed power storage is provided to handle peak loads of relay switching without reflection into the primary winding.

b. Fixed Sequencer Mechanization

The fixed sequencer, as shown in Figure 8G-3, provides redundant maneuver capability within the CC&S. The computer and sequencer are electrically independent except at their output interface where voting on the output function is performed with quad redundancy. Thus maneuver operations are protected from single component failures. Prior to the start of a fixed sequencer maneuver, the duration values for spacecraft pitch, rolls, and motor burn must be received via ground command.

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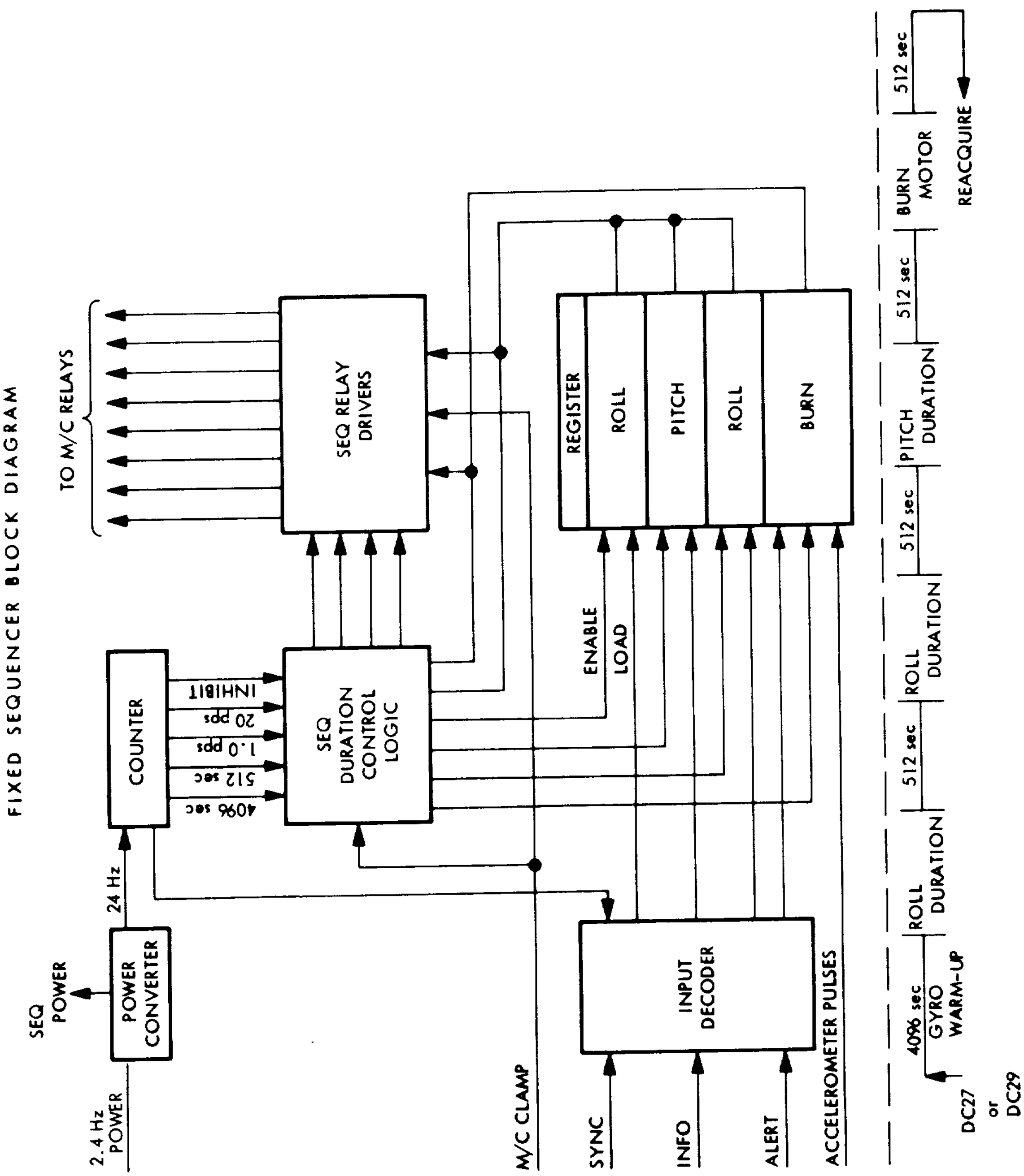

Figure 8G-3. Block Diagram of Sequencer

Subsequent receipt of DC-27 will immediately start the maneuver by removing the maneuver clamp which allows the counter to count. Similarly, the computer has been mechanized to start the maneuver. 4096 seconds after receipt of DC-27 or a computer enable signal, the counter overflows and starts the roll turn for the duration stored. After completion of roll, a fixed interval of 512 seconds is counted which, at its conclusion, starts pitch for the stored duration. After completion of pitch, another 512 seconds later, the second roll turn is executed. At completion of roll, another fixed interval of 512 seconds is counted before starting motor burn for the stored duration. 512 seconds after motor burn completion, the spacecraft goes through turn unwinds -- roll, pitch, roll. At a fixed interval of 512 seconds after unwind maneuver completion, the Attitude Control Subsystem is returned to the cruise mode and automatic reacquisition starts immediately. Pitch and roll turn durations are counted down at a 1-pps rate and may have values from 1 to 2047 seconds. The motor burn duration is counted down by the output pulses from the accelerometer. An abort of the maneuver via DC-13 will immediately reset the fixed sequencer and allow a second (fixed sequencer) maneuver to be executed as soon as proper roll, pitch, roll, and motor burn durations have been established. The fixed sequencer consisted of the following functional units:

1) Input decoder
2) Counter
3) Duration control logic
4) Storage registers
5) Relay drivers
6) Power conversion

(1) Sequencer Input Decoder

The input decoder provides the following functions: selects sequencer load words from CC command words, transfers appropriate sections of the word to the proper magnetic shift register for storage, checks the load word for proper parity, and stores pitch and roll polarity until needed in executing the maneuver.

(2) Sequencer Counter

It is the function of the sequencer counter to provide those timing signals necessary to perform a fixed sequence midcourse maneuver. To this end the counter accepts a 2.4-Hz input reference frequency and divides it down to provide signals to operate the turn registers (motor burn duration is provided by counting accelerometer pulses), to provide a fixed delay between maneuver events and to provide the initial delay period required for gyro warmup.

(3) Sequencer Duration Control Logic

The heart of the duration control logic is a magnetic core stepping switch which controls the counter gating noted in the previous section, controls turn on of roll, pitch, roll, and motor burn registers in the count mode, and controls initiation of fixed time event relay drivers to activate maneuver events.

(4) Sequencer Storage Registers

The sequencer storage registers used to store and execute maneuver turn and motor burn durations are magnetic core shift registers of 11 bits each. Each register is provided with exclusive OR logic feedback from the ninth and eleventh stages in order to generate $2^{11}-1$ discrete count cycle for duration timing. (The motor burn duration register is buffered by ÷32 counter.) It must be noted that the count cycle is not binary and is best determined by referring to tables listing each sequential discrete state (table for counting state is shown in PD 605-139, "Mariner Mars 1969 Spacecraft Central Computer and Sequencer Coded Command Synthesis").

(5) Sequencer Relay Drivers

Included in the fixed sequencer are relay drivers and a sink driver that provides pulsed continuity. These drivers are independent of the computer in order to guarantee that a single component failure in the relay drivers will not affect the ability of the computer to actuate maneuver event relays.

(6) Sequencer Power Conversion

Power conversion for the sequencer is relatively simple in comparison to that required by the computer since only two voltages, +4 VDC and +28 VDC, are needed. These DC voltages are turn-on surge limited, have no regulation, and are DC isolated from the spacecraft 2.4-KHz input power by their transformer-full-wave rectifier generation system.

3. Baseline Functional Considerations

The basic configuration of the CC&S for Viking is essentially the same as that of Mariner'71. Sequencing for launch/cruise, midcourse, orbit insertion, capsule checkout initiation, capsule release initiation, and orbital operations would be provided with CC&S updating or reprogramming in flight. The addition of a capsule-lander on the spacecraft requires additional CC&S subroutines. It is the purpose of this section of the report to delineate the requirements of the Viking CC&S design.

a. Memory Capacity Considerations

The most important part of a programmable CC&S is its memory capacity. It is in the memory that the mission sequence of events is stored. Therefore, the reliable use of the CC&S in any flight mission can only occur if adequate memory capacity exists to store basic subroutines such that ground command reprogramming is held to a minimum. The allocation of memory for the various mission event sequences for the Viking mission is as follows:

(1) Launch/Cruise

During the launch-cruise phase of the mission, the CC&S is required to generate a sequence of events for countdown, spacecraft separation, attitude stabilization, and cyclic updating and testing. It is anticipated that the Viking CC&S program will require 38 memory locations for launch/cruise subroutines.

(2) Midcourse Maneuver(s)

During the midcourse maneuver(s) the CC&S is required to initiate the turn on of the gyros, to provide timing for the proper spacecraft attitude orientation, and to provide counting for motor burn termination. Presently, Viking requires three commanded turns - roll, pitch, and roll, for engine and maneuver antenna orientation. Because stray light in the vicinity of the planet affects both the sun sensor and the Canopus sensor, performing a reverse attitude maneuver (unwind maneuver) after engine burns is desirable. The Viking midcourse maneuver program requires 37 memory locations.

(3) Antenna Pointing

The CC&S provides updating commands for the high gain antenna throughout the Viking mission. These commands contain antenna position and periodic times for updating the antenna. The simplest CC&S mechanization is to use a "cyclic-pulse event" to update the antenna. Upon occurrence of the cyclic event, a series of clock pulses are sent by the CC&S to the antenna position control mechanism to drive the antenna to the desired position. At present it is assumed that the antenna control subsystem has only one degree of freedom. Cyclic event pulses are provided on a daily basis. However, if finer pointing of the high-gain antenna is required, a technique similar to the pointing of the scan platform would be employed.

(4) Scan Platform Pointing

The Viking orbiter will require a two degree-of-freedom (clock and cone) scan platform. The scan platform and control utilize the MM'71 design. Studies have indicated that a near optimum selection

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of minimum CC&S memory and best pointing accuracy is to use a linear-slope approximation technique. The preliminary CC&S memory requirements for linear-slope programming indicate that 41 words are required for one slope with an allowance for the return of the platform to an initial position. Additional slopes require 9 words each for a change in one axis and 15 words for a change in both axes. The scan platform pointing sequence is initiated by the central computer and sequencer signal which turns the scan platform on. The "power on" signal commands the scan to slew the platform to the required clock and cone angle references. After some preset time, which allows for ground verification of scan performance, the CC&S then commands the platform to point at the prescribed clock and cone angle. The pointing is accomplished by a series of clock reference update pulses from the CC&S to slew the platform to the required position at the proper times.

(5) Orbit Insertion

During the orbit insertion phase, the CC&S provides the spacecraft with a sequence of events such that the gyros are turned on, the motor gimbal angle null positions are updated (if necessary), the spacecraft attitude is rotated to the proper orientation, and the motor burns for the proper velocity increment so that the predetermined orbit can be achieved. The execution of the sequence will be totally automatic at a predetermined orbit insertion time. The orbit insertion program will be initiated by the computer with ground command as a backup. The Viking orbit insertion CC&S program consists of 47 memory locations assuming no solar occultation contiguous to orbit insertion. Solar occultation backup switching will be handled by the CC&S during orbital operation.

(6) Orbital Operation

Prior to the release of the lander-capsule, a series of scientific experiments will be conducted by the orbiter. The CC&S presently plans to provide power on/off commands for each of the experiments on board the orbiter. The CC&S will also control the pointing of the scan platform and the playback rate of the telemetry data. Because of stray light effects from the planet during orbital operation, the CC&S will act as a back-up for the stray light sensor to place the spacecraft in a roll inertial mode. It is anticipated that during the Viking orbital operation the spacecraft will experience solar occultation. During these periods the CC&S will provide backup switching to place the spacecraft into inertial hold. During orbit trim the CC&S will provide the trim control functions. One of the prime requirements of the orbiter is to act as a communications relay link between the lander and Earth. The CC&S will provide commands to control the tape recorders and to point the antenna. Preliminary estimates indicated that orbital operations will require a minimum of 168 words of memory.

(a) Capsule-Lander Checkout

One of the capsule requirements on the orbiter system is that the Viking orbiter CC&S has the ability to initiate the capsule-lander, checkout program. The assumptions made are that the capsule-lander sequencer has a memory capacity of at least 64 locations and that the checkout program requires 46 memory locations.

Option A. The capsule-lander checkout program will be stored in the lander sequencer memory. The lander mission sequence will be stored in the lander. The capsule-lander checkout program will be initiated by the orbiter's CC&S. Any updating or reprogramming of the lander sequencer can be handled through the orbiter command system via the CC&S. It is anticipated that a subroutine requiring 12 CC&S memory locations will be required. Two alternatives to the above method are described in options B and C.

Option B. The capsule-lander checkout program will be performed by the orbiter CC&S via the LC&S. The orbiter CC&S will allocate 46 memory locations to store the capsule-lander checkout program. The checkout program will consist of turning on various science subsystems in the lander, a time verification of various event occurrences and updating the lander sequencer. The mission sequence of the capsule lander will be stored in the lander sequencer's 64 memory locations. Any variation of the lander mission sequence will be accomplished by reprogramming the orbiter

CC&S, then a transfer in storage locations between the orbiter CC&S and the capsule-lander sequencer.

Option C. The assumption made in this option is that the capsule-lander checkout program is initially stored in the lander sequencer memory. To initiate the lander sequencer checkout program would require 4 CC&S memory locations. After the completion of the lander checkout, the lander's mission sequence program stored in the CC&S memory will be transferred to the lander-sequencer memory. Hence, the total number of CC&S memory locations required to satisy the option C requirement will be 74 CC&S locations.

(b) Capsule-Lander Separation Maneuver

The pre-capsule lander maneuver requires the spacecraft attitude to be changed to provide the proper orientation for the capsule deflection motor firing. This maneuver is essentially a no-burn midcourse. Therefore, no additional CC&S memory is required. However, two CC&S memory locations will probably be required to enable and back up the separation of the lander from the orbiter. After the ejection of the lander, an unwind maneuver will be required to permit reacquisition of the celestial references.

Table 8G-1 summarizes the central computer and sequencer memory requirements for the Viking mission for various lander checkout options.

b. Maneuvers

The Viking spacecraft requires many maneuvers throughout its mission. There are essentially two types of maneuvers: spacecraft attitude maneuvers and maneuvers which require motor burn. The CC&S can control these maneuvers in three different modes: tandem, computer only, and sequencer only. The tandem mode is the maneuver control mode in which the fixed sequencer is the prime maneuver element with the computer programmed to follow along and check its operation. The computer only mode is the control mode where spacecraft maneuvers are controlled only by the

TABLE 8 G-1

Mission Phase	No. of CC&S Memory Locations		
	Option A	Option B	Option C
Launch/Cruise	38	38	38
Midcourse Maneuver	37	37	37
Orbit Insertion	57	57	57
Orbital Operation	168	168	168
Capsule-Lander Checkout	12	58	74
Capsule-Lander Ejection	10	10	10
Antenna Pointing*	4-50	4-50	4-50
TOTAL	326-372	372-418	388-434

*The smaller number represents antenna pointing by generation of a cyclic event.

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computer. During the sequencer only mode the spacecraft is controlled only by the sequencer. Table 8G-2 summarizes the CC&S maneuver control modes.

(1) Midcourse Maneuver Considerations

The Viking mission will use the sequencer to integrate the output accelerometer pulses and the computer as a timed backup for control of midcourse burn (tandem mode). As indicated above, in the tandem mode the fixed sequencer is the prime maneuver element with the computer programmed to follow along and check its operation. Should the sequencer err, or should the computer err in checking the sequencer during the turns, the maneuver would be automatically aborted. However, the checking process is modified for the motor-burn event. To allow the motor to start, the computer issues a start signal just prior to the expected sequencer start. If the sequencer start occurs within 2 seconds of the computer start, the motor is fired, the abort is disabled, and both the computer and sequencer simultaneously start to count down the motor-burn duration. Whichever completes the countdown first issues the stop-motor-burn command. This provides backup protection against an overlong burn and increases the probability of an early shutdown, which could be corrected by a second midcourse maneuver.

(2) Orbit Insertion Burn Maneuver Consideration

The orbit insertion phase of the Viking mission can be ground command independent. Therefore, the execution of the orbit insertion burn maneuver should require no ground commands. The CC&S maneuver control mode selected for orbit insertion is the sequencer only mode. The sequencer is initiated by the computer at a preprogrammed nominal time. The computer then acts as a backup for the orbit insertion burn maneuver.

4. Interface Definitions

The CC&S provides services to the various subsystems of the orbiter and the lander. These services are in the form of commands to activate various mission events and signals for control of the spacecraft.

Table 8G-2. Maneuver Mode

CC&S MANEUVER CONTROL MODES

MANEUVER PERIODS	NORMAL AND BACKUP MANEUVER CONTROL MODES						REMARKS
	TANDEM		COMPUTER ONLY		SEQUENCER ONLY		
	TURNS (TIMED)	BURN (COUNTS ACCEL./ TIMED)	TURNS (TIMED)	BURN (TIMED)	TURNS (TIMED)	BURN (COUNTS ACCEL.)	
MIDCOURSE MANEUVER	NORMAL	NORMAL	BACKUP	BACKUP	BACKUP	BACKUP	3 TURN MANEUVER
RELEASE OF BC AND VL.	NORMAL		BACKUP		BACKUP		
ORBIT INSERTION			BACKUP	BACKUP	NORMAL	NORMAL	3 TURN MANEUVER
ORBIT TRIM	NORMAL	NORMAL	BACKUP	BACKUP	BACKUP	BACKUP	2 TURN MANEUVER
TURN UNWINDS (USED FOR ALL MANEUVERS)			NORMAL				BACKUP BY COMMANDED REACQUISITION

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Almost all communication between the orbiter and the lander is accomplished via the orbiter CC&S. The functional interfaces between the CC&S and the various subsystems (including the lander) are presented below.

a. CC&S-Orbiter Subsystems

(1) Command Subsystem

The CC&S receives all data via the command subsystem interface.

(2) Telemetry Subsystem

The interface between the CC&S and the telemetry subsystem is essentially that of switching functions, i.e., switching of data modes, bit rates, etc.

(3) Radio Subsystem

The CC&S provides to the radio subsystem switching functions such as switch maneuver antenna, switch beacon transmitter off, etc.

(4) Power Subsystem

The CC&S controls the distribution of power to the various subsystems and controls the charging of the spacecraft battery via switching functions.

(5) Attitude Control Subsystem

The CC&S controls: (a) all attitude maneuvers of the spacecraft, (b) positioning of the scan platform, (c) updating of the motor gimbal angles, and (d) pointing of the high gain antenna drive mechanism. The CC&S provides various control functions for the attitude control subsystem, such as, gyro and autopilot power switching.

(6) Pyro Subsystem

The CC&S acts as a backup to the L/V-S/C Separation Initialed Timer for the firing of squibs during the separation phase of the launch. The CC&S also controls the burning of the rocket motor during midcourse and orbit insertion maneuvers. In addition to the above, the CC&S provides switching functions to deploy solar panels, pressurize the system, open propellant lines, etc.

(7) Science Subsystems

The CC&S provides power switching for the various scientific instruments.

b. CC&S-Lander

Since the CC&S will handle almost all of the communication between the orbiter and the lander, the CC&S will be equipped with a transfer buffer (including a parity generator) and a number of status flag flip-flops. The buffer and control flip-flops provide data and control links to and from the CC&S and lander computer and sequencer. All quantitative data or updated data for the lander computer will be handled by the CC&S.

H. FLIGHT TELEMETRY SUBSYSTEM

The flight telemetry subsystem (FTS) is responsible for performing the conditioning, encoding, multiplexing, and subcarrier modulation of engineering data and the subcarrier modulation and block coding of scientific data.

A block diagram of the FTS is given in Figure 8H-1. The engineering measurements are commutated in ten decks sampled at four rates. One deck consists of ten 7-bit words. A whole data frame (at least one sample of every measurement) requires 14 min at 33 1/3 bps or 56 min at 8 1/3 bps. Frame synchronization and subcommutation indexing are provided by words in the highest speed deck. The engineering data biphase-modulates a 24-KHz subcarrier.

Low rate science data at 133 1/3 bps is in the form of a serial bit stream from the SDS and is modulated on to a 34.286-KHz subcarrier in the FTS.

The third telemetry channel is high rate science data from the data storage subsystem. This is 16.2, 8.1, 4, 2 or 1 Kbps data which is block coded with a (32, 6) comma free code and modulated on a 259.2-KHz subcarrier in the FTS.

The spacecraft engineering channel operates continuously during the entire mission. Either the low or high rate science is turned on during various portions of the mission. Capsule data at 1348 bps during pre-separation check out, separation, and entry will be conditioned and replace the low-rate science channel on the 34.286-KHz subcarrier. Capsule low-rate engineering data (1050 bph) will be subcommutated as one 7-bit word on the high-rate deck of the engineering commutator. The high rate deck is read out every 4.2 seconds at 33 1/3 bps or every 16.8 seconds at 8 1/3 bps.

FTS Parameters:

Weight	= 26 lbs
Power	= 15 watts
Bit Rates	= 33 1/3 bps, 8 1/3 bps Engineering
	= 133 bps Low Rate Science
	= 16.2 Kbps, 8.1 Kbps, 4.05 Kbps, 2.025 Kbps, 1.0125 Kbps Block Coded Science

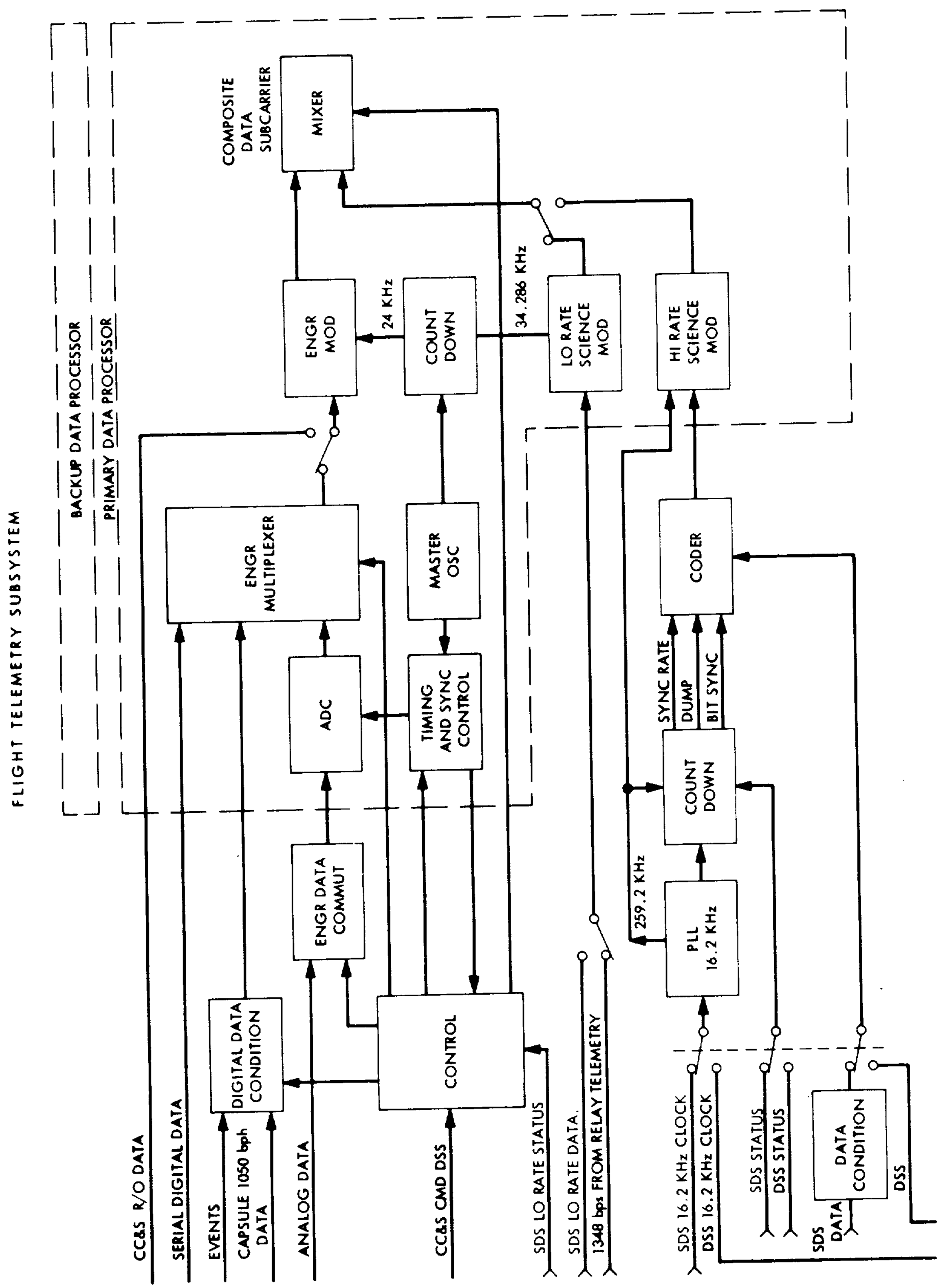

Figure 8H-1. Flight Telemetry Block Diagram

Subcarrier = 24 KHz Engineering
34.286 KHz Low Rate Science
259.2 KHz Block Coded Science

I. FLIGHT CONTROL SUBSYSTEM

1. Introduction

The flight control subsystem consists of the attitude control, thrust vector control, and antenna pointing control subsystem.

During all phases of the mission, except during motor burns, three-axis control of the spacecraft attitude is provided by the attitude control system. This control system is made up of the reaction control system (cold N_2 gas system), gyros, celestial sensors, and associated electronics and logic. This control system performs the following functions:

1) Removal of initial spacecraft tumbling rates which occur at launch vehicle/spacecraft separation.
2) Acquisition of celestial references with capability for automatic reacquisitions as required.
3) Maintenance of stable limit cycle behavior during periods of transit and orbital cruise.
4) Provide inertial hold capability for commanded turns and Sun occultations.
5) Provide 3rd axis (roll) control during the motor burn phase of spacecraft maneuvers.
6) Removal of spacecraft tumbling rates imparted at capsule separation.

During the motor burn phase of the midcourse, orbit insertion and orbit trim maneuvers, attitude control is provided by the thrust vector control system (autopilot) and the roll channel of the attitude control system. This control system points the engine thrust vector through the spacecraft center of mass and maintains pitch and yaw attitude stability. The roll channel of the attitude control system is required for roll stability.

During the late transit phase and during the orbital phase, the high gain antenna pointing direction is controlled by the antenna pointing control system. This control system, using CC&S commands and Earth-based updates, maintains the antenna pointing direction towards Earth.

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The following is a subsection description of the Viking Mars 1973 baseline flight control subsystem:

1) Functional Description--Describes the operation of the flight control system during each phase of the mission. The differences from the MM71 flight control subsystem are noted.
2) Mechanization Description--Lists the hardware that constitutes the flight control subsystem. An explanation of the function of each component is provided along with the required modification to the MM71 flight control subsystem.
3) Guidance and Control Analyses--Provides a discussion of the reaction control gas system sizing analysis, the thrust vector control system analysis and the guidance accuracy error analysis.
4) Weight and Power Summary--Describes the weight and power requirements of the flight control subsystem.
5) Additional Considerations--Provides a discussion of the different propulsion, thrust vector control and reaction control system configurations that were examined during the study. In addition, problems and areas requiring further study are listed.

2. Functional Description

The following functional description presents a discussion of the operation and performance of the various elements of the flight control subsystem during each phase of the mission.

a. Prelaunch

The prelaunch operations are limited to the following three items:

1) All system logic is reset at spacecraft power application.

2) Canopus cone angle is selected.

3) Gyro heading data is recorded and evaluated.

b. Launch

During the launch phase the gyros are on and caged in the rate mode. The accelerometer is on and electrically captured. The pitch and yaw gimbal servos are on and in a closed loop configuration. The Canopus sensor power is off, but the Sun shutter power is on due to the lack of a Sun gate signal. The attitude control system is inhibited since there is no power to the switching amplifier preamplifiers.

c. Sun Acquisition

At separation the control system is automatically set into the Sun acquisition mode. All logic resets to the Sun acquisition mode to safeguard against inadvertent changes in the logic element states which can result from the launch environment. At the completion of Sun acquisition, the Sun gate issues a signal which places the system in the Canopus acquisition mode. Canopus acquisition however is not initiated until either a CC&S command is issued or a direct command is received by the spacecraft.

d. Canopus Acquisition

Upon receipt of either a CC&S or direct command, the Canopus sensor power is turned on and the roll search generator is enabled. This command causes the Canopus sensor to perform a "fly-back and sweep" operation prior to enabling of the roll search generator. This operation insures acquisition of Canopus if it is in the Canopus sensor field of view at the time of power turn-on. The roll search inhibit circuit is also enabled to safeguard against spacecraft "spin-up" in the event of a failure in the Canopus sensor (saturated output) or the roll gyro. At the conclusion of a normal acquisition, the Canopus acquisition signal turns the gyros off (after a 180-sec delay) and places the system in a cruise mode of operation. In the event that there is an anomaly in the Canopus sensor intensity logic which prevents acquisition of <u>any</u> star, a direct command has been mechanized to switch the switching amplifier input to the Canopus sensor and disable the intensity gate logic. This will enable the system to acquire any star within the Canopus sensor sensitivity threshold.

e. Transit Cruise

During the cruise mode the spacecraft is controlled by celestial references (Sun and Canopus). The gyros are off and the gas jet reaction control system is controlled by the derived rate switching amplifiers. The spacecraft roll axis will be aligned with the Sun within ±0.25 degrees and the Canopus sensor boresight axis will be aligned with Canopus within ±0.25 degrees.

f. Midcourse Maneuver

The midcourse maneuver sequence is different from the MM71 and previous Mariner spacecraft. The two primary differences are:

1) Three commanded turns may be necessary in order to guarantee that the maneuver antenna is in the correct position to communicate with Earth. The order of commanded turns will be roll, pitch, roll.
2) Sun and Canopus reacquisition after motor burn will be accomplished by unwinding the commanded turns. Two options are available for unwinding the commanded turns. The first is to repeat the turns in reverse order and polarity. The second method unwinds the turns using a two-turn roll-pitch sequence. This method provides faster reacquisition of the references but requires the transmission of two additional quantitative commands from the ground. Unwinding of turns is in contrast to Mariner spacecraft, which allow automatic Sun reacquisition using the primary and secondary Sun sensors and Canopus reacquisition by initiating a roll search until Canopus falls into the field of view.

The sequence of events for midcourse maneuver is as follows. The inertial reference unit (gyros and accelerometer) is commanded on by CC&S at the start of the maneuver. The gyros are in the rate mode and are turned on for a 68-minute warm up prior to the commanded turns. The spacecraft remains on celestial references during this period. Shortly after the gyros are turned on, the CC&S will transfer the motor gimbal angles to the pitch and yaw pre-aim circuits (pre-aim capability may be required if CG migration is severe) which include digital-to-analog converters in the autopilot electronics. This is necessary in order to point the motor thrust vector close to the spacecraft center of gravity. Telemetry of the motor gimbal position will verify that the gimbals have moved to the correct position. The next event after the 68 minute gyro warm-up and motor gimbal pre-aim is the start of the roll turn. At this time the roll gyro is commanded to inertial hold, the turn polarity is commanded and the commanded turn generator in the inertial reference unit is enabled. The pitch and yaw gyros remain in rate mode and the spacecraft roll axis remains Sun oriented under control of the Sun sensors. After the roll turn is completed the pitch and yaw gyros are commanded to the inertial mode and the pitch turn magnitude and polarity are commanded by the CC&S. The Sun sensors are turned off and the spacecraft attitude is controlled by the gyros in the inertial mode. At the end of the pitch turn, a second roll turn may be necessary in order to orient the maneuver antenna.

The next event is motor burn. At the start of motor burn (if path guidance is used), the pitch and yaw path guidance amplifiers are switched into the autopilot preamplifier feedback loops, the roll gas jet deadband is opened up, and the pitch and yaw switching amplifiers are disabled in order to conserve nitrogen gas during motor burn. During the motor burn, roll control is maintained by the roll gas jets. The length of the motor burn is controlled by a counter in the CC&S which counts pulses from the unidirectional accelerometer in the inertial reference unit. The scale factor of the accelerometer is 0.03 meters/sec/pulse. At the end of motor burn the autopilot gimbal servo drive electronics and the path guidance amplifiers are disabled.

In order to reacquire the Sun and Canopus, CC&S commands the spacecraft to unwind the turns. After the turns are completed the spacecraft will be Sun and Canopus oriented. The CC&S then releases the gyros from the inertial mode and places the attitude control system back into the acquisition mode. The Canopus tracker is turned on and the autopilot electronics are turned off. The attitude control system then acquires the Sun and Canopus and then returns to the cruise configuration. If a second midcourse maneuver is required, the above sequence of events is repeated in the same manner.

g. Orbit Insertion Maneuver

The orbit insertion maneuver will be performed in an identical sequential manner to the midcourse maneuver described in detail above. A simplified sequence of events is listed below.

1) Turn on inertial reference unit for gyro warm-up.
2) Transfer the motor gimbal angles from CC&S to the autopilot pre-aim circuitry (if pre-aim is required).
3) Perform a roll turn.
4) Perform a pitch turn.
5) Perform a second roll turn in order to orient the maneuver antenna toward the Earth.
6) Initiate motor burn.
7) Terminate motor burn.
8) Unwind roll turn.
9) Unwind pitch turn.
10) Unwind roll turn

The spacecraft is now Sun and Canopus oriented.

h. Post Orbit Insertion/Pre-capsule Separation

After orbit insertion, the spacecraft is in orbit for approximately ten days before the capsule separation phase is initiated. A discussion of the operations during this phase are given below.

(1) Cruise

After completion of the orbit insertion burn and unwinding of the commanded turns, the spacecraft is Sun and Canopus oriented and operating in the cruise mode. Cruise mode operation during this phase is identical to the transit cruise operations discussed previously.

(2) Stray Light Operations

During orbit, if the lighted limb of the planet appears within the field of view of the stray light sensor (corresponding to the Canopus tracker interference region) the roll gyro is automatically placed in the rate mode. Three minutes later the roll gyro is automatically commanded to inertial hold. At this time the Canopus error signal from the Canopus sensor is disabled and the spacecraft roll axis is controlled instead by the roll gyro. As long as the stray light condition exists the roll axis remains inertially controlled. At the end of the stray light condition, the Canopus sensor error signal is enabled and the roll gyro is switched to the rate mode. Three minutes after the loss of the stray light signal, the roll gyro is turned off and the spacecraft is returned to the normal "orbit cruise" mode.

(3) Commanded Turns

During the ten day period between orbit insertion and capsule release, several sets of commanded turns will be required for orbit trim maneuvers, science orientation, bioshield cap jettison, and capsule orientation. These commanded turns, including their unwinding, are performed in the manner previously described for the midcourse maneuver. The gyros are commanded on for a 68 minute warm-up. The roll gyro is then set to the inertial mode and the roll turn is performed. The pitch and yaw gyros are then commanded to inertial mode and the pitch turn is performed. After the correct spacecraft attitude is reached and the purpose for performing the turns has been accomplished, the attitude control system unwinds the turns. Celestial references are then reestablished by the reacquisition of the Sun and Canopus and the spacecraft returns to the cruise mode.

(4) Orbit Trims

One or more orbit trim maneuvers will be required for orbital corrections in preparation for capsule release. These trim maneuvers will typically require very small ΔV corrections and therefore very small motor burn durations. Thrust vector and attitude control during the orbit trim maneuvers is accomplished in the manner described previously for the mid-course maneuver. Depending on the CG shifts that may have occurred during the orbit insertion burn, gimbal pre-aiming may be required.

i. Capsule Separation Phase

The attitude control system has four major functions to perform during the capsule separation phase of the Viking mission:

1) Perform spacecraft turn for bioshield cap jettison.
2) Perform spacecraft turns for capsule release.
3) Remove spacecraft rates imparted at capsule and bioshield cap separation.
4) Remove spacecraft rates imparted at bioshield base jettison.

The operations for the commanded turns are described in 2.h.(3) above. The third function listed above is particularly important. At capsule release, the attitude control system is in inertial hold. Immediately after capsule separation, the attitude control system must remove any imparted spacecraft rates before excessive position errors result which might saturate the gyro integrators. This requirement places a constraint on the capsule release mechanism in terms of allowed spacecraft tip-off rates. For example, if the maximum allowable position error offset about each of the spacecraft axes is 5 degrees, then the spacecraft tip-off rates cannot exceed 1 deg/sec. about pitch and yaw, and 0.1 deg/sec about roll axis.

j. Orbit Cruise

After completion of the capsule separation phase, the spacecraft remains in orbit for at least an additional 90 days. During this orbit phase the attitude control system is in the same configuration and operates in the same manner as it is during the transit cruise phase.

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k. Sun Occultations

Depending on which orbit is selected, the spacecraft will begin to enter Sun occultation, once each orbit, as soon as 50 days and as late as 87 days after orbit insertion. The Sun occultation periods will vary from a few minutes initially and increase to approximately two hours by the end of the mission. To accommodate this condition, the Viking spacecraft will fly a newly designed electro-optical device to sense impending Sun occultations. This device, a Sun occultation sensor, along with the Sun gate will provide the switching necessary to maintain spacecraft attitude reference during Sun occultations.

The sequence is as follows: The lighted limb of the planet, falling in the field of view of the Sun occultation sensor, will cause a signal to be sent to the attitude control electronics to turn on the gyros. The gyros will remain on and in the rate mode until the spacecraft begins to disappear behind the planet as viewed from the Sun. As the Sun's intensity diminishes at the beginning of occultation,the Sun gate changes state and sends a signal to the attitude control electronics to place the spacecraft in inertial hold. At this time the Sun sensor and Canopus sensor error signals are disabled and the spacecraft attitude is controlled by the gyros. This mode continues throughout the Sun occultation period. At the end of Sun occultation, the Sun gate will change state commanding the gyros to the rate mode and simultaneously enabling the Sun sensor and Canopus sensor error signals. The spacecraft is again under celestial control (Sun and Canopus) and,after a three minute delay, the gyros are automatically commanded off and the spacecraft returns to the normal "orbit cruise" mode.

l. Orbit Trim Manuevers

Orbit trim maneuvers during this phase of the mission are conducted periodically to correct and adjust the orbit. They will consist of a roll-pitch-roll commanded turn sequence, a short motor burn, and unwinding of the commanded turns. The operations during this phase are identical to the midcourse maneuvers during the transit phase.

3. Mechanization Description

The flight control subsystem mechanization consists of the following:

1) Attitude control electronics and logic.
2) Reaction control system (cold N_2 gas system).
3) Celestial sensors (Canopus tracker, Sun sensors, Sun gate, stray light sensor, and Sun occultation sensor).
4) Inertial reference unit (gyro and accelerometer assembly).
5) Thrust vector control system (autopilot).
6) Antenna pointing control system.

A block diagram of the flight control subsystem is shown in Figure 8I-1.

a. Attitude Control Electronics and Logic

The attitude control electronics consists of the following:

1) Analog switching amplifiers (3-axes) to control the nitrogen gas solenoid valves.
2) Sun gate electronics.
3) Canopus gate electronics.
4) Stray light sensor electronics.
5) Gyro time delay.
6) Logic circuits for mode switching.

The attitude control electronics will be essentially the same design as for MM71. Some additional logic circuits will be necessary to mechanize the Sun occultation mode.

A Sun occultation sensor, Sun gate, and the CC&S will provide signal inputs to the new logic electronics. In addition, an extended time

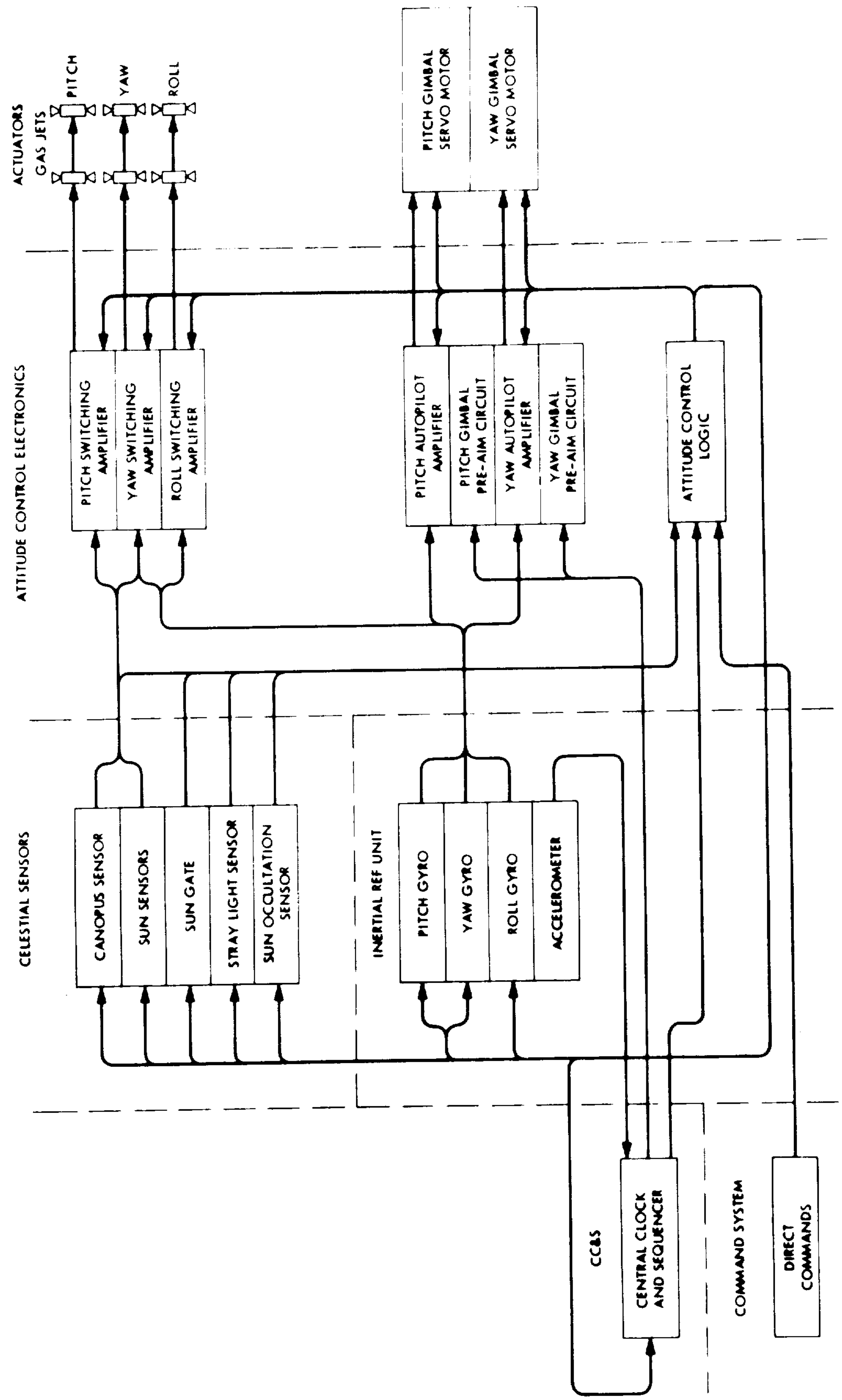

Figure 8I-1. Flight Control Subsystem Block Diagram delay circuit will be needed to keep the gyros on during the period between the occurrence of the Sun occultation sensor signal and the actual Sun occultation.

b. Reaction Control System

The reaction control system (A/C gas system) consists of the following components:

1) High pressure gas tanks (2)
2) Pressure regulators, 15 psia. (2)
3) Nitrogen pressure transducers (2)
4) High pressure and low pressure gas distribution system
5) Pitch, yaw, and roll valve manifolds and valves

The reaction control system is a dual redundant gas system of the type that has been used on Mariner '64, Mariner '67, Mariner '69 and is proposed for Mariner '71. Control torques about each axis are provided by valve couples, thereby eliminating cross coupling between axes. A diagram showing the location of valves is given in Figure 8I-2. By virtue of its dual redundant nature, it can automatically compensate for a valve failure (no ground command is required). The initial gas storage weight for this type of system is determined from the half-gas-system consumption for the mission. Half-gas-system operation occurs if one valve sticks open. Under these conditions, two-thirds of the stored gas is lost, leaving one-third to complete the mission. Therefore, the initial gas storage weight is set at three times the half-gas-system requirement. An increase in the reaction control system weight, from previous Mariner spacecraft, will be necessary for the Viking spacecraft. This is due to increased gas consumption. The increase is due to the larger vehicle inertias, requiring higher jet thrust levels, and the larger number of discrete events such as commanded turns. The gas storage requirement, and how it is determined, is discussed further in subsection 4.a. below.

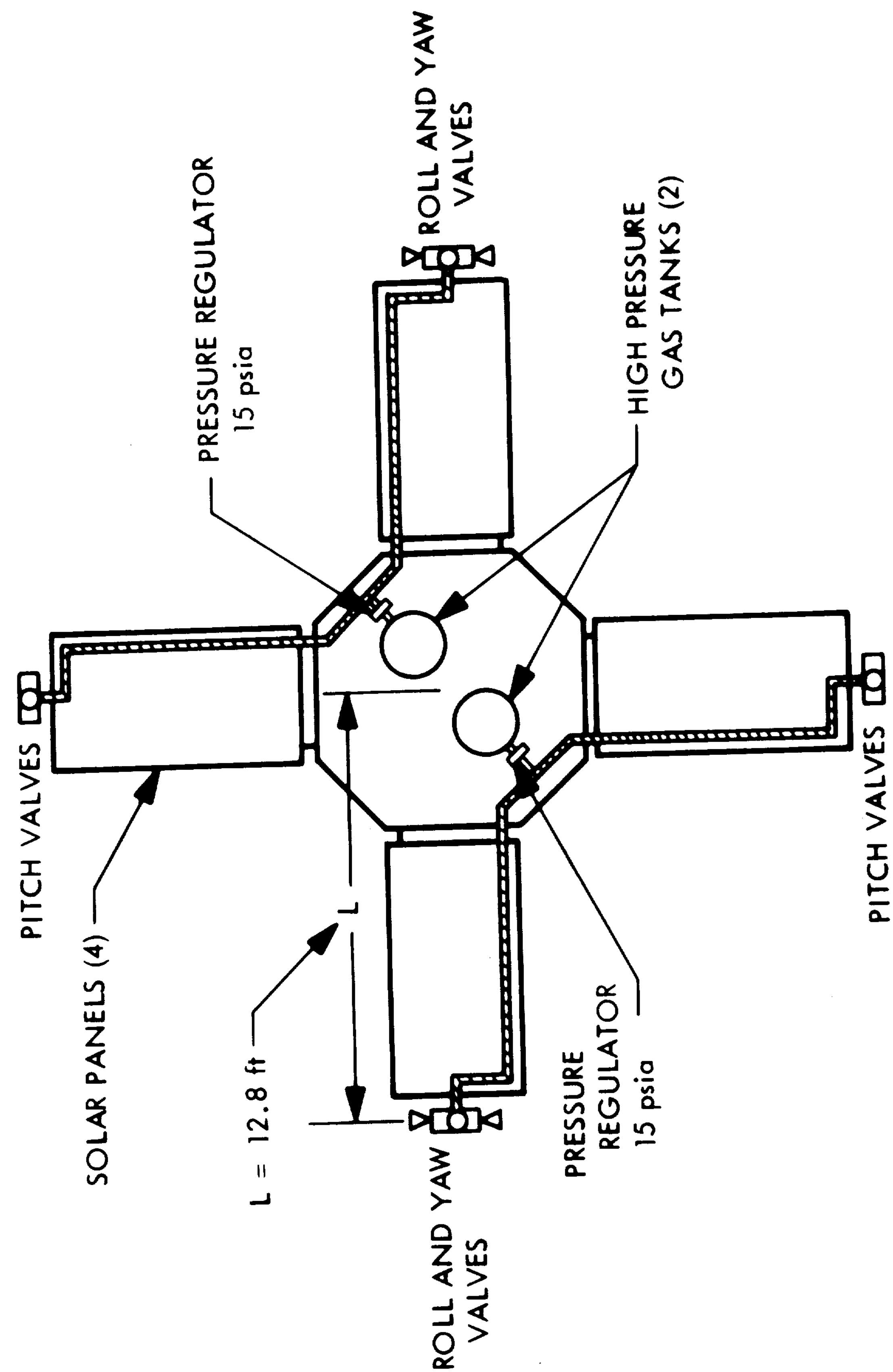

Figure 8I-2. Reaction Control System Schematic Diagram

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c. Celestial Sensors

The celestial sensors on the Viking spacecraft consist of the following:

1) Canopus tracker and Sun shutter assembly
2) Primary Sun sensors
3) Secondary Sun sensors
4) Sun gate assembly
5) Stray light sensor
6) Sun occultation sensor

(1) Canopus Tracker

The Canopus tracker will be similar to that used on the Mariner '71 mission. No special problems concerning the use of the Mariner '71 Canopus tracker are foreseen because of stray light or Canopus occultation. The proposed orbital geometry for the Viking mission will produce less severe stray light conditions than the Mariner '71 mission because of its lower orbital inclination.

The field of view of the Canopus tracker is overlapped by the field of view of the stray light sensor. If the lighted limb of the planet appears within the stray light field of view, the stray light sensor produces a signal which places the roll channel in inertial hold until the stray light condition has ended. This condition is displayed in Figure 8I-3 for the planet coordination system shown in Figure 8I-4.

(2) Primary Sun Sensors

A single primary Sun sensor assembly will be used similar to the assembly to be used on MM71. The modular assembly will contain both the cruise sensors and the acquisition sensors for the pitch and yaw axes.

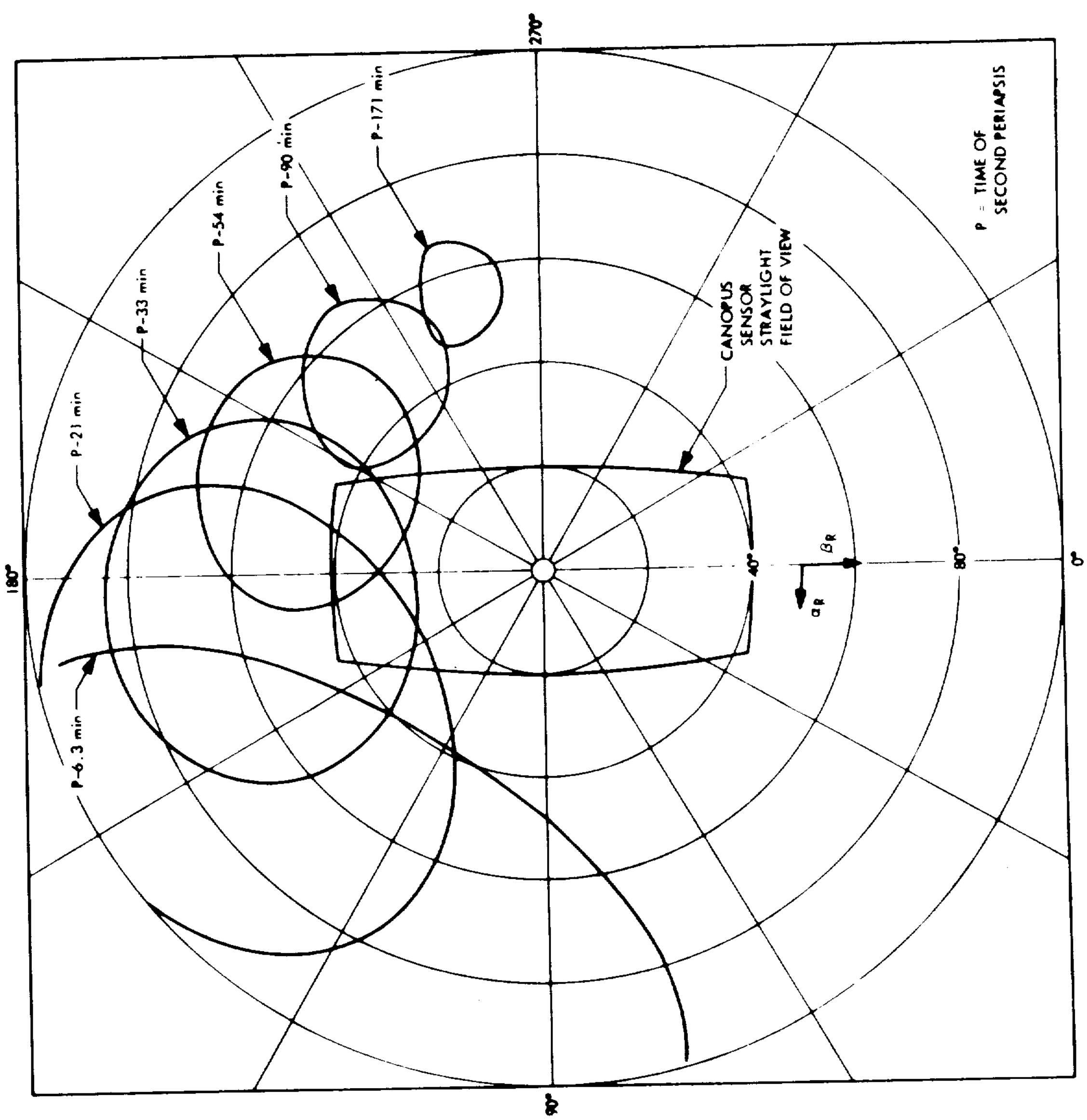

Figure 8I-3. Canopus Tracker Straylight Study

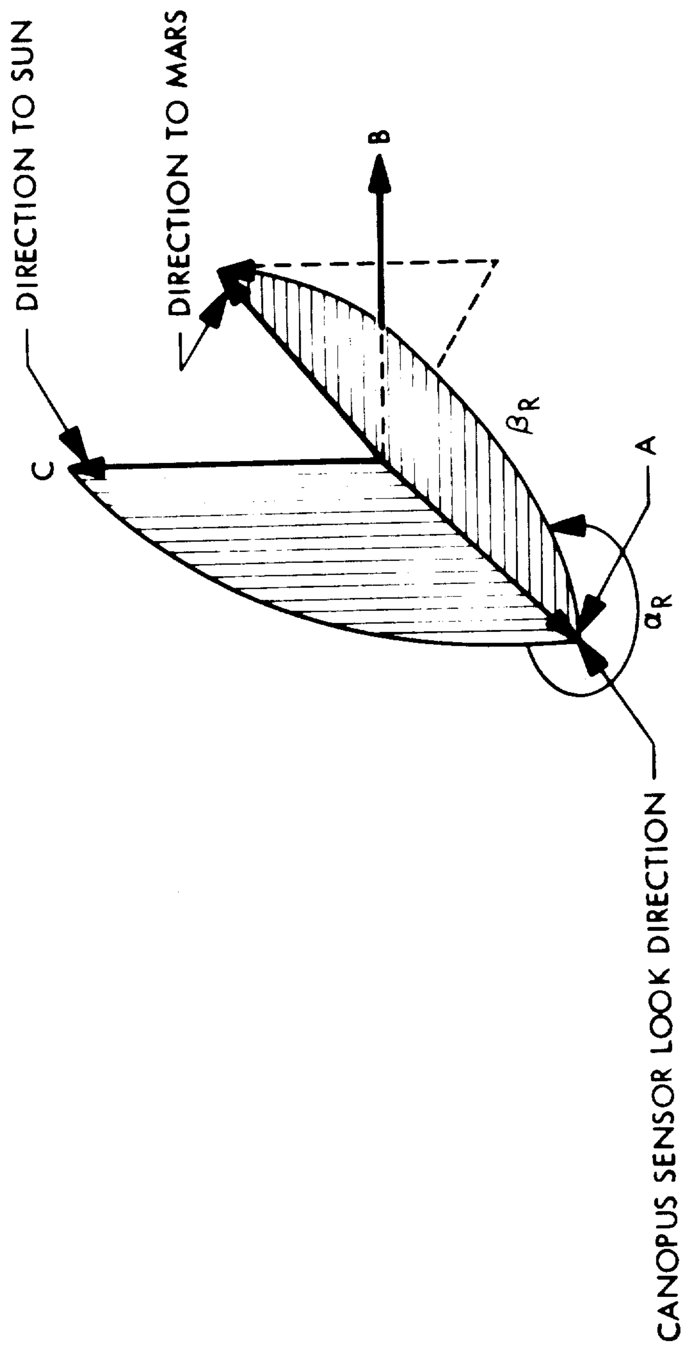

Figure 8I-4. Planet Geometry Coordinate System

(3) Secondary Sun Sensors

Four secondary Sun sensors may be relocated from the bus structure to the solar panels. In order to achieve an unobstructed field of view, it may be necessary to repackage the secondary sensors into two assemblies located on opposite solar panels.

(4) Sun Gate

The Sun gate is equivalent to the unit on MM71 and will be mounted near the primary sensor assembly. Minor mechanical changes are expected to accommodate its relocation.

(5) Stray Light Sensor

The stray light sensor is an electro-optical device that has the same bore-sight axis as the Canopus sensor. Its field of view is larger than the field of view of the Canopus sensor. Its purpose is to sense the presence of stray light in the vicinity of the Canopus tracker. Stray light may cause the Canopus tracker to exhibit an error offset thereby producing a pointing error in the direction of Canopus. The Mariner '71 stray light sensor is planned for use on the Viking mission without modification.

(6) Sun Occultation Sensor

The Sun occultation sensor for the Viking mission will be a newly designed electro-optical device. At present two designs are being considered by the Celestial Sensors Design Group. The first is a slit sensor using the design technique of the Mariner '69 narrow-angle Mars gate, and the second is a conical sensor using the design technique of the Mariner '71 stray light sensor.

A brief description of the two designs is given below.

(a) Slit Sensor

This sensor contains a fan shaped slit oriented in the pitch-yaw plane of the spacecraft. Its function is to sense the lighted limb of the planet and generate a signal when the planet passes through the sensor slit and approaches the Sun. This signal turns on the gyros for the necessary gyro warm-up. Since the slit sensor has a narrow field of view in the orbital plane, the planet will be in view for only a short time period. This requires that an electronic memory must be included in the attitude control electronics to keep the gyros on during the period between slit sensor actuation and Sun occultation. The Sun gate can be used as a switching device to sense actual Sun occultation and to command the gyros to inertial control. The CC&S will be used to enable the Sun occultation mode.

(b) Conical Sensor

This sensor will have a conical field of view and point along the Sun line. The center of the conical view area which points at the Sun will be masked off. Also, the portion of conical view area that is not useful in determining impending Sun occultation will be masked off. The sensor will observe the lighted limb of the planet when the limb is within approximately 15 degrees of the Sun direction. This will initiate gyro turn-on. The continued presence of the Sun within the sensitive area of the conical sensor will keep the gyros on. As Sun occultation nears, the lighted limb of the planet becomes smaller and eventually disappears from the field of view of the conical sensor prior to the occultation. This requires a short electronic time delay in order to keep the gyro power on until actual Sun occultation occurs. As in the previous sensor design, the CC&S will enable the Sun occultation mode and the Sun gate will command the gyros to inertial hold.

d. Inertial Reference Unit

The inertial reference unit consists of two subassemblies: the inertial sensor subassembly and the inertial sensor electronic subassembly. The major components of these two subassemblies are listed below:

1) Pitch, yaw and roll gyros
2) Pitch, yaw and roll gyro electronics
3) Pitch, yaw and roll electronic integrators
4) Precision reference voltage supplies for the commanded turn generator, electronic integrators, accelerometer comparator, and accelerometer torque switch.
5) Mode switching and logic electronics
6) Roll incremental angle control electronics
7) Accelerometer
8) Accelerometer electronics

Figure 8I-5 shows a simplified block diagram of the inertial reference unit. This inertial reference unit is identical to the MM71 unit and can be used for the Viking mission without modification.

e. Thrust Vector Control System

Two basic options exist for mechanizing the thrust vector control system or autopilot. The choice of which mechanization to use will depend on the final determination of the predominant sources of thrust vector pointing error. The two basic mechanization options are listed below. Neither mechanization will require any major changes to the MM71 autopilot.

(1) Autopilot Using Path Guidance and Pre-aim

If the major source of pointing error is due to CG shifts and migration (x-y plane), then a path guidance autopilot, using pre-aim will provide the most accurate mechanization. Figure 8I-6 shows a single axis block diagram of this autopilot. The function of the path guidance circuit is to provide automatic compensation for CG offset errors and migration. This can be particularly critical for the long insertion burn. The function of the pre-aim circuitry is to bias the gimbal servo null point approximately through the CG prior to motor ignition. This eliminates large gimbal motion transients at motor ignition. In addition, the pre-aim reduces motor thrust vector to CG misalignments that could cause pointing errors during the short orbit trim burns.

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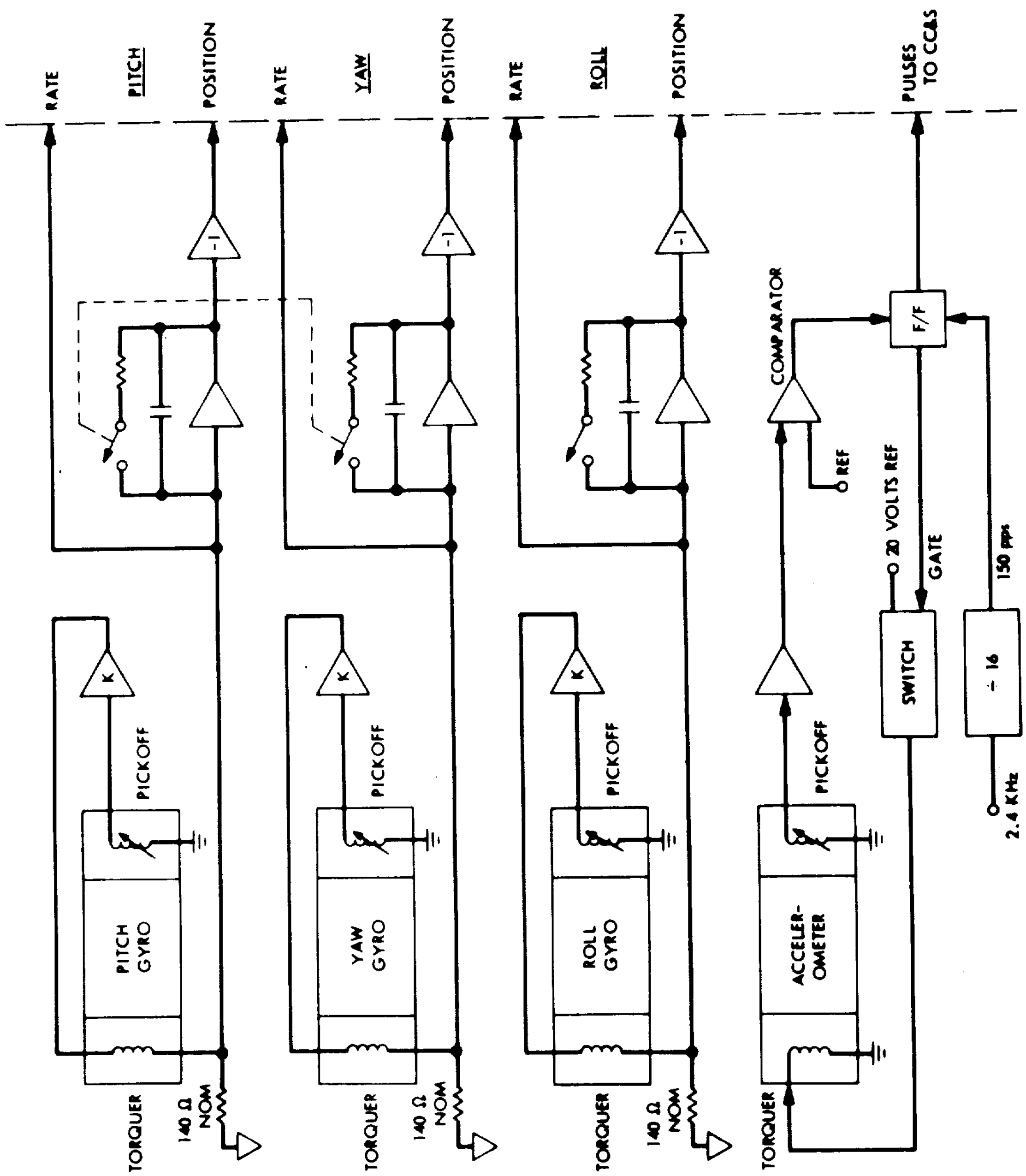

Figure 8I-5. Inertial Reference Unit Block Diagram

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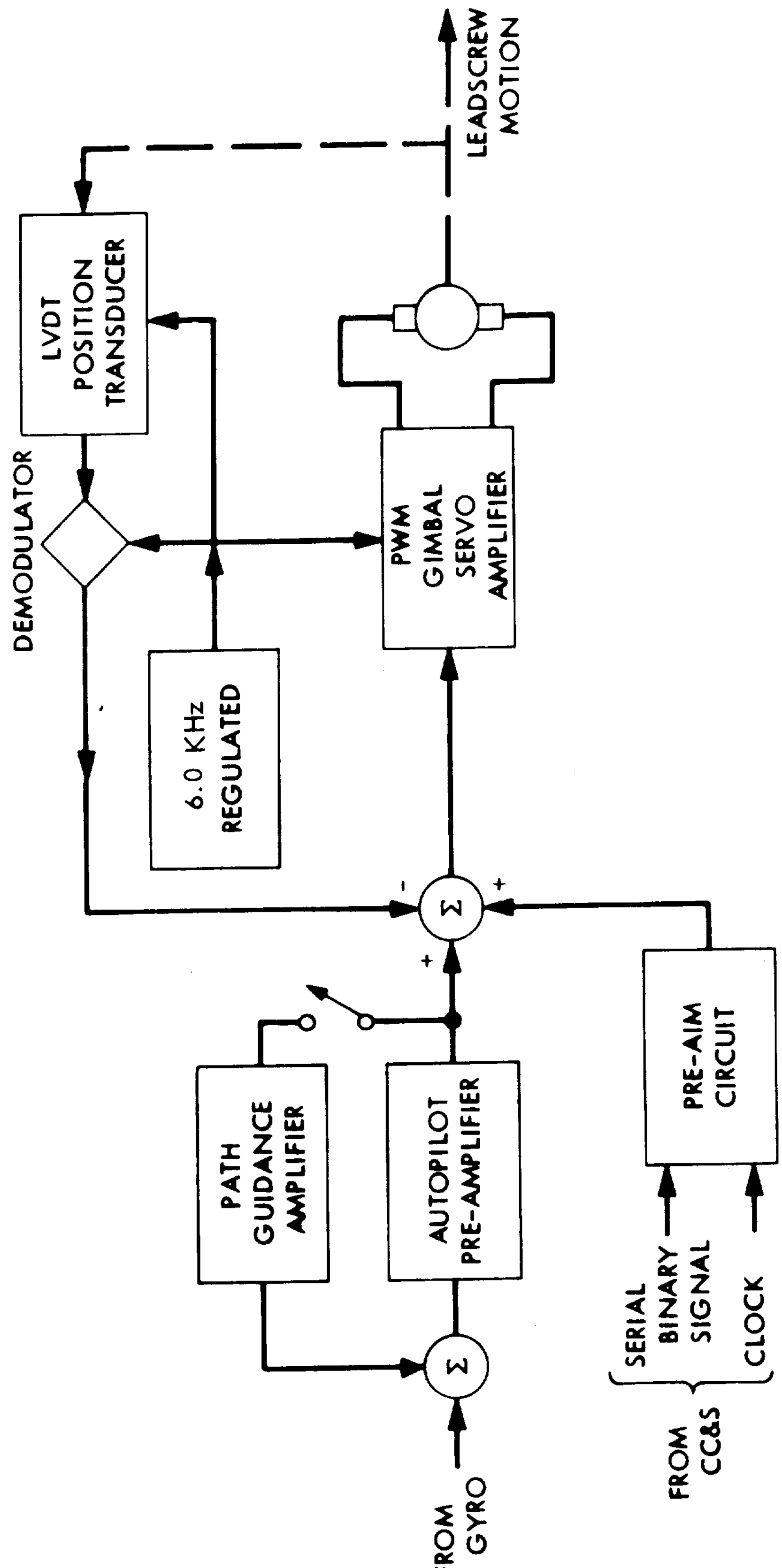

Figure 8I-6. Autopilot Block Diagram with Path Guidance and Pre-Aim

Operation of this autopilot option is discussed further in subsection 4.b. The autopilot electronics for this option consists of the following:

1) Pitch and yaw pre-amps
2) Pitch and yaw path guidance amplifiers
3) Motor gimbal pre-aim circuitry
4) Pulse width modulated gimbal servo amplifier
5) Linear variable differential transformer (LVDT)
6) 6.0 KHz regulated power supply

(2) High Gain Autopilot (lag-lead compensation)

A characteristic of the previous autopilot option is low DC gain. An inherent disadvantage of this is poor transient response. If CG offsets and migration are not the major source of pointing error, then this type of autopilot does not present any advantage over a conventional autopilot that does not utilize a path guidance loop. Sources of error, other than CG offsets, are the autopilot steady state error and engine misalignment errors. If these errors predominate, then an autopilot with high DC gain will provide the most accurate pointing. The basic autopilot and its electronics are identical to the previous option except that the path guidance loop is eliminated and amplifier circuitry is included for high gain compensation. This compensation scheme allows the autopilot loop gain to be set as high as 30 db and still have adequate stability margins. Pointing error for this mechanization is now strictly dependent on the magnitude of the CG offsets. Operation and performance of this mechanization are discussed further in subsection 4.b.

f. Antenna Pointing Control System

An antenna pointing control system will be required for orienting the high gain antenna from 30 to 40 days prior to orbit insertion through the remainder of the mission.

Analysis indicates that adequate pointing can be achieved using an appropriately oriented single degree-of-freedom antenna control system. A second degree-of-freedom is provided which deploys the antenna in a single step. The optimum location on the spacecraft for the single degree-of-freedom hinge axis and orientation of the antenna with respect to the hinge axis was determined through computer aided analysis. The optimization was performed over the period which begins at orbit insertion of trajectory "A" and extends to the end of the 100 day nominal mission of trajectory "B". The results of this analysis indicate that the optimum location of the hinge axis with respect to the spacecraft is

clock angle (θ) = 317.15 degrees, and

cone angle (ϕ) = 30.0 degrees.

The optimum antenna angle (ψ) is defined as the angle between the hinge axis and the antenna look direction and was found to be

antenna angle (ψ) = 25.83 degrees.

A second computer analysis was performed using the optimized angles. This analysis determined the best line segment fit to the hinge rotation angles that are required to track the Earth. The line segments were fitted from approximately 40 days prior to orbit insertion to somewhat beyond the nominal mission lifetime for each trajectory.

Figure 8I-7 and 8I-8 display the Earth position and antenna pointing position for trajectories A and B respectively. The difference between the two positions is due to two factors:

1) The antenna has only a single degree-of-freedom.
2) The hinge and antenna angles have been optimized for the orbital phase and therefore have somewhat greater error prior to orbit insertion than necessary.

This optimization period was specifically selected to ignore the pre-orbit insertion period since the range to Earth is relatively short at this time. This error in pointing is more than offset by the higher performance due to the short range.

The high gain antenna selected for the Viking Orbiter has a half beam width of 1.7 degrees at the 1-db point and 3 degrees at the 3-db point. The analysis indicated that two line segments provide an adequate fit for this antenna and for both trajectories.

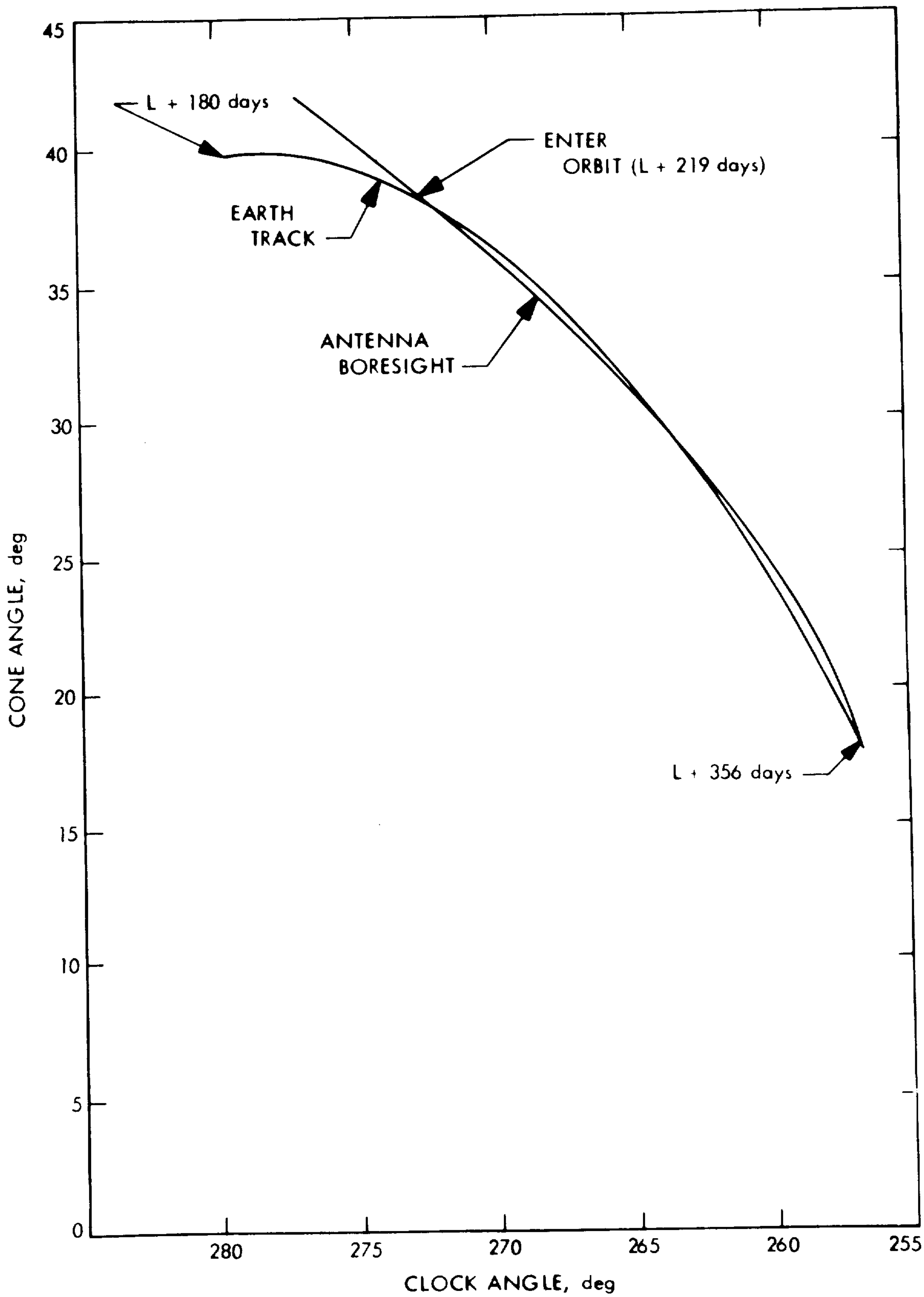

Figure 8I-7. Earth and Antenna Pointing Positions -- Trajectory A

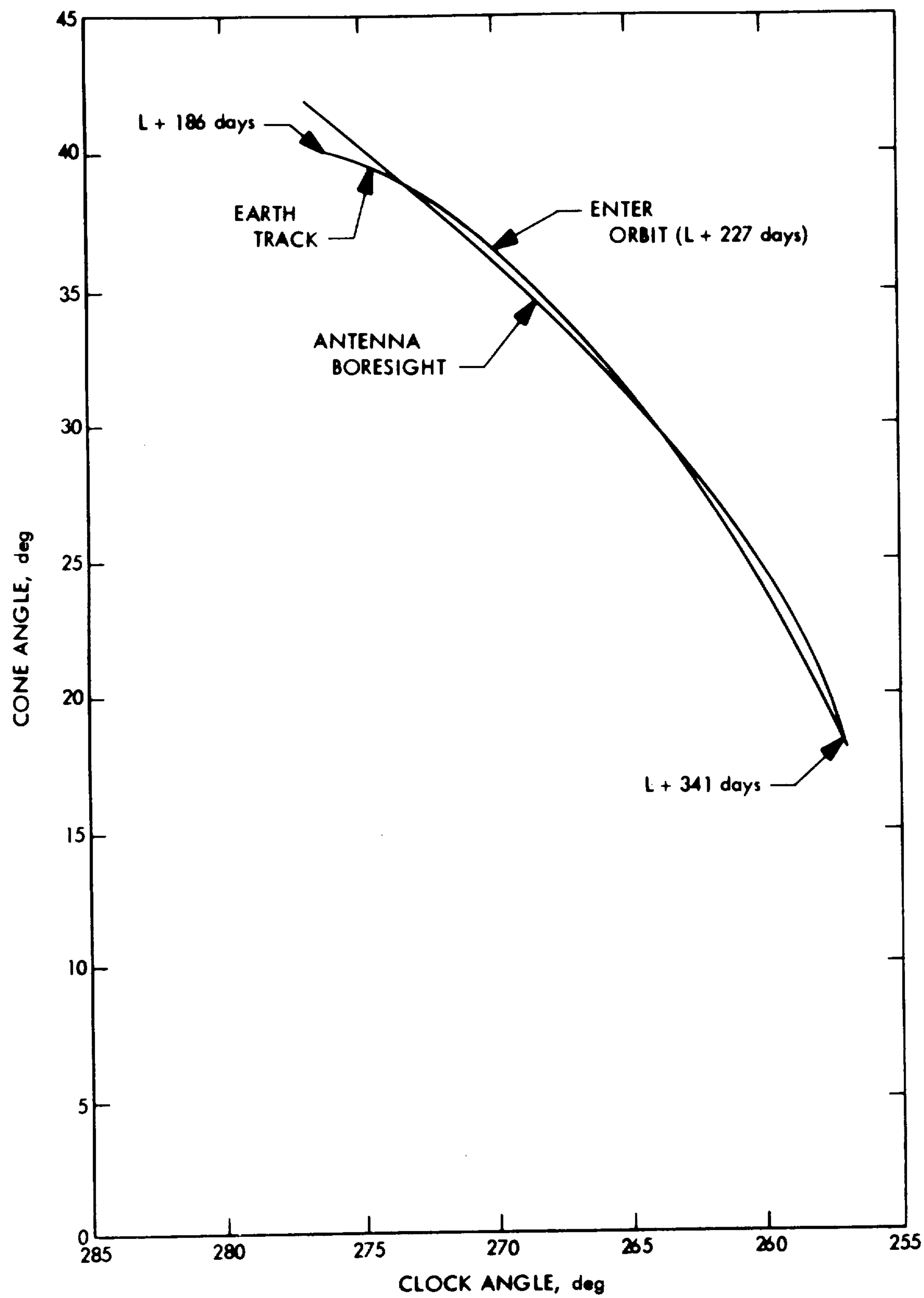

Figure 8I-8. Earth and Antenna Pointing Positions -- Trajectory B

Figures 8I-9 and 8I-10 illustrate the results of this for trajectories A and B respectively. Three curves are displayed on each plot. The first curve illustrates the geometric error due to single degree-of-freedom tracking and the selection of hinge and the selection of hinge and antenna angles. The second curve illustrates the effect of adding the line segment approximation errors. The third curve includes the effect of alignment, deadband and other spacecraft errors.

Several mechanization approaches are under consideration for the antenna pointing subsystem. The first approach utilizes a reference register, encoder or potentiometer that is stepped by CC&S or the FCS command. A closed loop stepper motor would be provided to drive the antenna. The second approach would utilize an open loop drive system again stepped by CC&S or command. The selection of the approach will depend primarily on relative complexity, accuracy and initial update considerations.

Regardless of the mechanization approach it appears that a daily "cyclic update" from CC&S will provide satisfactory accuracy. The update will be in the form of a pulse train. Two different length pulse trains, corresponding to the two line segments, are required. For the two trajectory cases considered, the slopes varied from about .25 degrees/day to 0.4 degrees/day.

4. Guidance and Control Analyses

a. Reaction Control System Analysis

An analysis of the reaction control system, and its gas storage requirements for the Viking mission, was made to determine the required gas storage and the overall RCS subsystem weight. To facilitate this analysis, and the many iterations that were necessary, a digital computer program was written for the IBM 7094 computer. The features of this program are listed below.

1) Computes gas consumption for separation rate removal, acquisitions, searches, overrides, commanded turns (including unwinding), and roll axis gas consumption during motor burns.

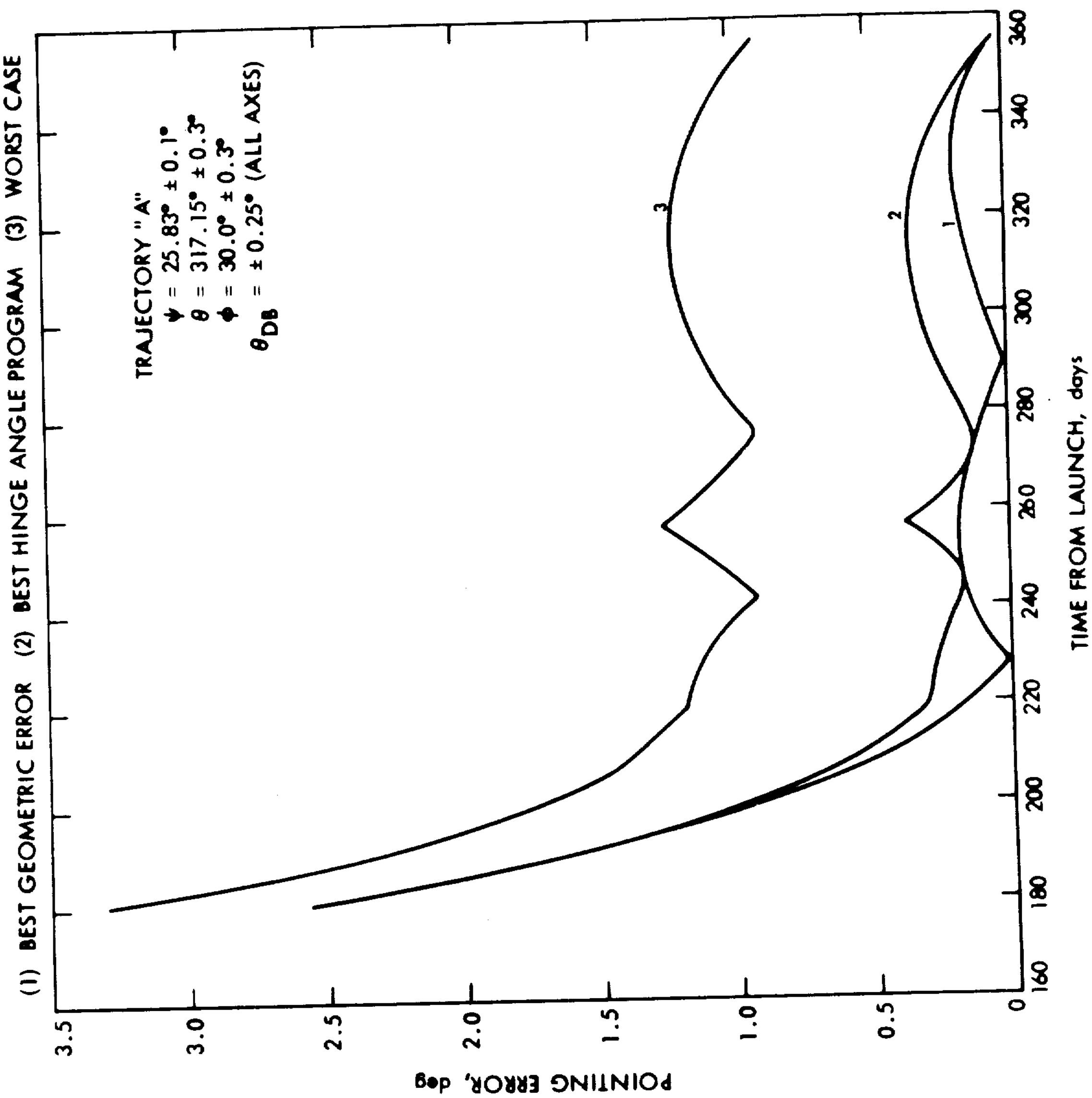

Figure 8I-9. Antenna Pointing Errors Plots -- Trajectory A

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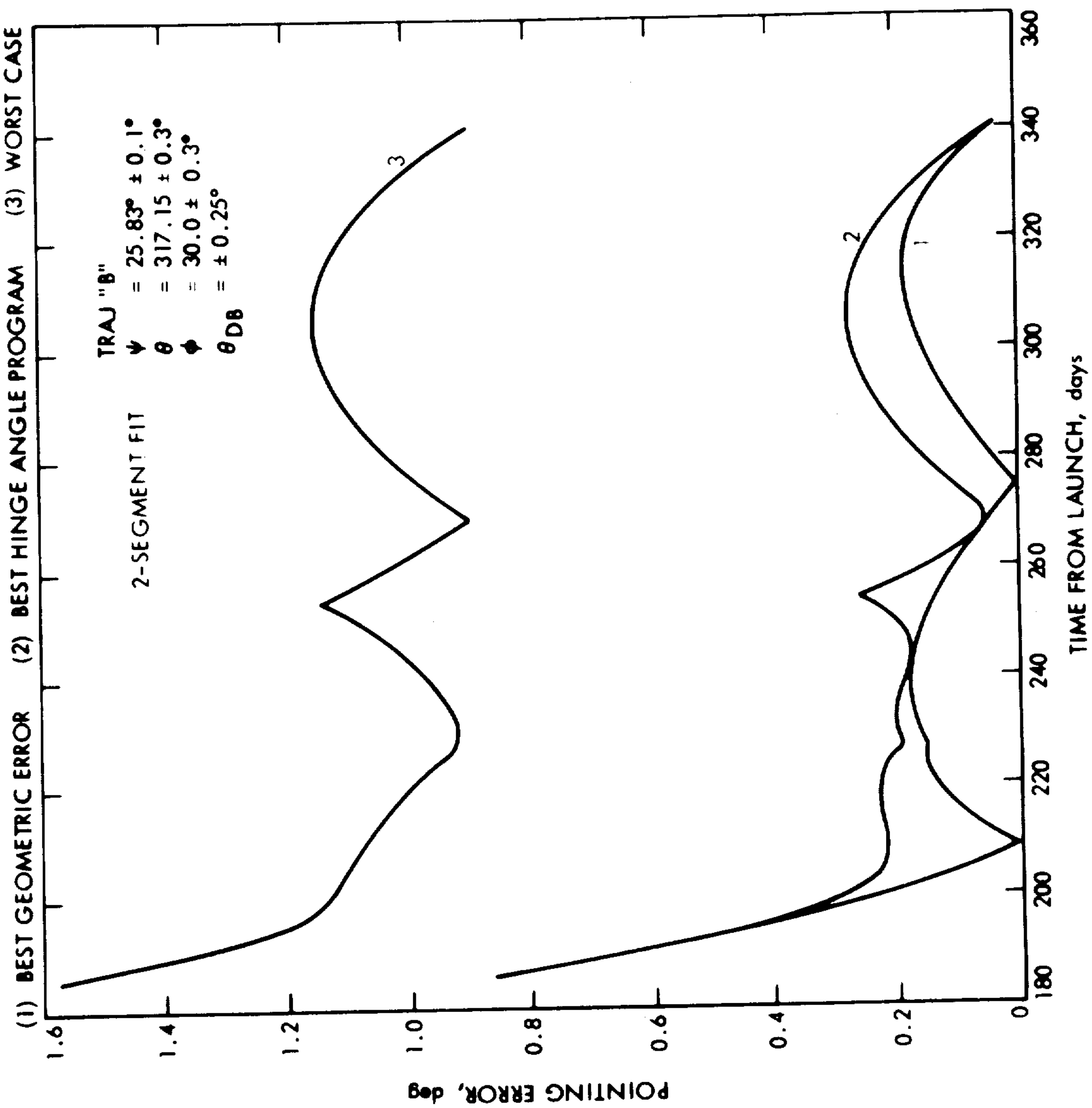

Figure 8I-10. Antenna Pointing Error Plots -- Trajectory B

2) Computes limit cycle gas consumption, both ideal and when under the influence of disturbance torques (solar, gravity gradient, etc.).
3) Computes gas consumption per day throughout mission.
4) Computes gas consumption for each phase of the mission.
5) Provides for variation in moments of inertia and attitude control gas system parameters.
6) Provides for simulation of failure modes such as a valve-open-failure (half system failure).
7) Computes total gas consumption during mission.

Through the use of this computer program, the gas consumption and gas system weights shown in Table 8I-1 were determined. The gas storage weights are based on the following conditions during the mission:

1) Initial separation rate removal - 3 deg/sec (each axis)
2) Capsule separation rate removal - 1 deg/sec, pitch and yaw, 0.1 deg/sec, roll
3) 6 roll searches
4) 20 roll overrides
5) 6 acquisitions (Sun and Canopus)
6) 2 sets of roll-pitch-roll commanded turns for midcourse maneuvers (including unwinds).
7) Total motor burn duration for all maneuvers ≈ 45 min.
8) 1 set of roll-pitch-roll commanded turns for orbit insertion with unwinds after completion of burn.
9) Roll-pitch-roll commanded turns (including unwinds) after orbit insertion and before capsule release:
   a) 1 set of turns for an orbit trim
   b) 8 sets of turns for science orientation
   c) 1 set of turns for bioshield cap release
10) 1 set of roll-pitch-roll commanded turns for capsule orientation with unwinds after capsule release
11) 4 sets of roll-pitch-roll commanded turns with unwinds for orbit trims after capsule release

Table 8I-1. Gas Consumption Weight Summary*

	FULL GAS SYSTEM		HALF GAS SYSTEM	
	DAILY USAGE lb/day	GAS WEIGHT lb	DAILY USAGE lb/day	GAS WEIGHT lb
● CRUISE				
LIMIT CYCLING:				
TRANSIT (220 days)	0.00356	0.783	0.000803	0.177
ORBITAL (100 days)	0.0217	2.17	0.00542	0.542
SUN OCCULTATIONS	0.00295	0.295	0.000738	0.0738
GRAVITY GRADIENT	negligible	-----	negligible	-----
LEAKAGE	0.00377	1.206	0.001890	0.605
SUBTOTAL		4.454		1.399
● DISCRETE EVENTS				
INITIAL RATE REDUCTION (3 deg/sec)		0.614		0.614
CAPSULE SEPARATION RATE REDUCTION		0.0244		0.0244
ACQUISITIONS (6)		0.192		0.192
ROLL SEARCHES (6)		0.0799		0.0799
ROLL OVERRIDES (20)		0.129		0.129
COMMANDED TURNS		1.778		2.026
MOTOR BURNS		0.518		0.518
SUBTOTAL		3.335		3.583
TOTAL		7.789		4.982

*RCS sized for three times half gas system requirement - 14.95 lb

12) Transit cruise period - 220 days
13) Orbital cruise period - 100 days
14) Continuous disturbance torque: pitch and yaw - 50 dyne cm, and roll - 10 dyne cm.
15) Gas system leakage rate - 30 cc/hr/half system

The gas storage requirements are summarized in Table 8I-1. The large increase in gas storage weight over that required for previous Mariner spacecraft (5.2 lb) is due to the much larger vehicle inertias and increased number of discrete events. The vehicle inertias for the Viking spacecraft and their changes during the mission are given in Table 8I-2. The vehicle inertias prior to capsule separation are over an order of magnitude greater than the inertias of any previous Mariner spacecraft. For a fixed jet thrust level, the limit cycle gas consumption increases as the vehicle inertias decrease. In addition, more gas is required for performing the large number of discrete events. In particular, the total number of commanded turns is greatly increased. Gas consumption just for turns is 2.026 lbs and is the greatest contributor of all the sources of gas consumption.

The selection of the proper gas jet thrust level is based on a trade-off between the following:

1) Limit cycle behavior
2) Vehicle response with regard to rate removal, commanded turns, and acquisition of celestial references
3) Removal of vehicle rates imparted at capsule separation. This is particularly important because the spacecraft attitude control system is in inertial hold at capsule separation. If the vehicle rates are greater than approximately 1 deg/sec about pitch and yaw and 0.1 deg/sec about roll, excessively large attitude errors will occur. In addition, saturation of the gyro electronic integrators, which derive position error information, can occur.

Table 8I-2. Spacecraft Inertia and Engine Lever Arm Summary

PARAMETER \ VEHICLE CONFIGURATION	MIDCOURSE AND BEGINNING OF ORBIT INSERTION (220 days)	POST-ORBIT INSERTION PRE-CAPSULE SEPARATION (10 days)	POST-CAPSULE SEPARATION (90 days)
PITCH AXIS INERTIA (slug ft^2)	3864	2536	585.7
YAW AXIS INERTIA (slug ft^2)	3758	2502	551.3
ROLL AXIS INERTIA (slug ft^2)	1654	1238	803.0
MOTOR LEVER ARM (ft)	4.11	5.97	2.56

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Suitable values for the jet thrust levels are
pitch and yaw jets - .07 lbs, and
roll jets - .014 lbs.

These thrust levels represent typical values and are a compromise between the above three constraints.

b. Thrust Vector Control System Analysis

The autopilot or thrust vector control system provides attitude control during engine burns for the midcourse maneuvers, orbit insertion maneuver, and orbit trims. This is accomplished by mounting the engine in a gimbal system as shown in Figure 8I-11. The engine is the long burn duration Rocketdyne RS-2101, 300 lbf, liquid bi-propellant engine. The engine and gimbal system are located on the z (roll) axis with the engine pointing in the -z direction. The gimbal control system positions the engine through the use of two linear actuators aligned with the pitch and yaw directions. This engine rotation capability allows the autopilot system to point the thrust vector through the spacecraft center of mass. With only this system, control cannot be effected about the z axis and, since swirl in the exhaust gases produces some torque, additional control must be provided for roll stability. This is accomplished using the cold gas roll attitude control channel.

As previously described in section 3. e., two basic autopilot mechanizations can be easily realized from the MM71 autopilot design. The results of the analysis and computer simulation of these two options are discussed below.

(1) Path Guidance Autopilot

If CG offset errors and migration are the most significant source of pointing error, an autopilot using path guidance will provide the most accurate thrust vector control.

A block diagram of this autopilot option is shown in Figure 8I-12. After completion of the commanded turns, which orient the spacecraft so that the engine is in the proper inertial attitude for the velocity change maneuver, a signal for motor ignition causes the motor burn switch to make two changes in the autopilot and attitude control system: (1) the pitch

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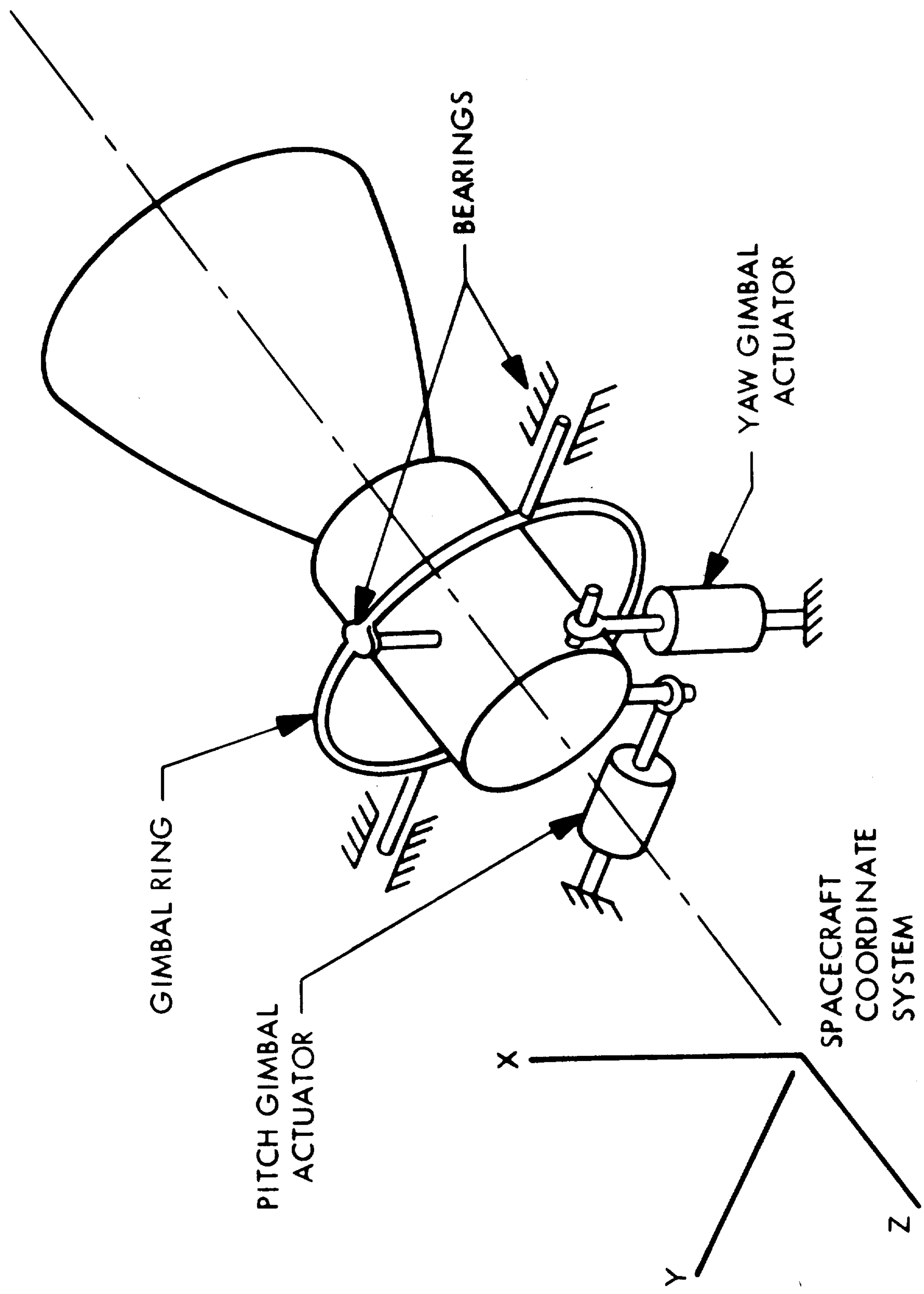

Figure 8I-11. Engine and Gimbal Mounting Arrangement

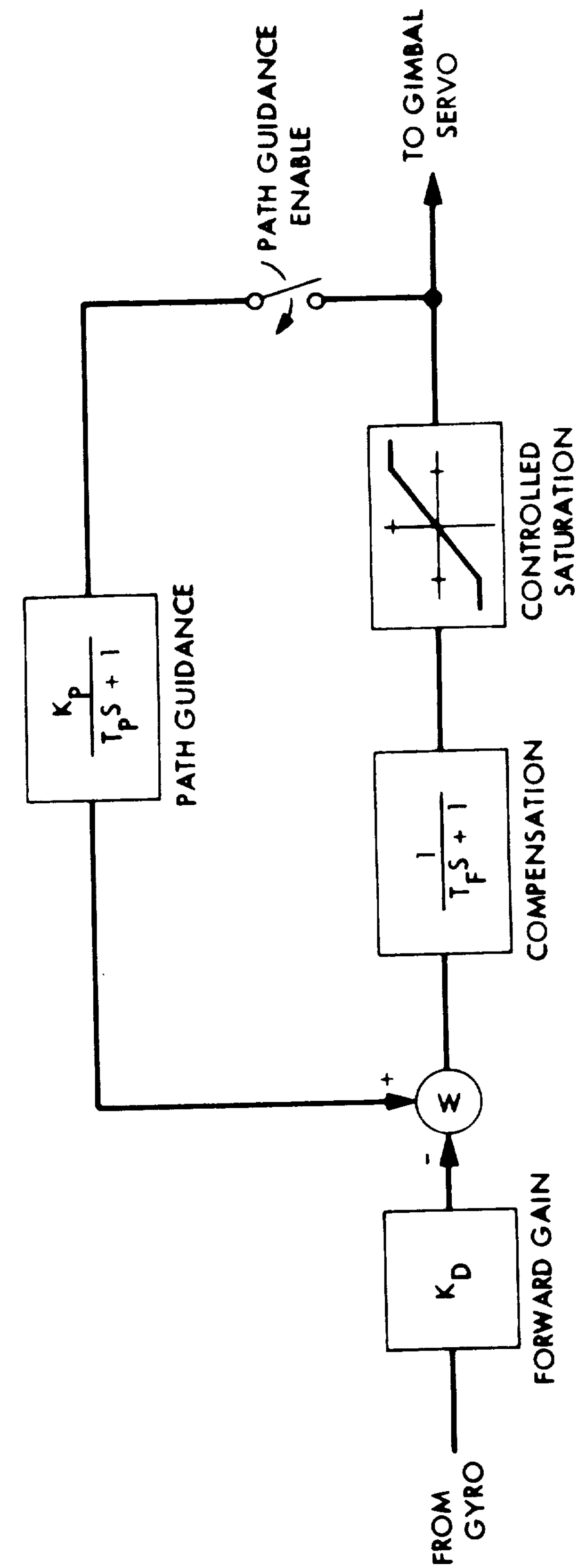

Figure 8I-12. Path Guidance Autopilot Block Diagram

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and yaw gas systems are disabled to prevent wasting large amounts of cold gas during the burn, and (2) the path guidance circuits are enabled. These circuits provide compensation for CG alignment erros and migration during the burn. The changes are completed simultaneously with motor ignition. At motor ignition, the spacecraft is in the position shown in Figure 8I-13. The commanded turns have been performed so that, excluding turn errors, the spacecraft z axis is aligned with the vector r, the inertial reference direction for the ΔV increment. The gimbal actuators are at null (or slightly off due to roll gas system limit cycling) and the engine is aligned with the calculated position of the CG. Due to errors in the CG determination, the calculated position does not coincide with the actual CG location. Figure 8I-14 shows the evolution of the thrust/CG relationship during the burn. Here the first sketch, showing the conditions at ignition, is exactly Figure 8I-13 with the spacecraft outline removed for clarity. This shows the thrust vector in the proper reference direction, but misaligned with the CG. This produces a negative torque on the spacecraft. The second sketch shows the resulting transient. As the torque rotates the spacecraft in the negative pitch direction, the pitch gyro senses the error, θ, between the reference direction r and the z axis, producing a positive output voltage. This voltage is filtered by the autopilot and used to drive the gimbal servo. The engine is now rotated until the angle ϕ is sufficient to pass the thrust vector through the CG. At this time the engine thrust vector is $\theta + \phi$. Without path guidance, the spacecraft would maintain this attitude during the entire burn. However, the path guidance circuit senses the gimbal command voltage and feeds it back to the autopilot input, through an appropriate filter, as shown in Figure 8I-12. This positive feedback causes the angle ϕ to increase very slowly in magnitude. The result on the spacecraft is a positive torque which slews the attitude as shown in the 3rd sketch in Figure 8I-14. The gains in the path guidance loop are set so that when the gyro error signal exactly cancels the positive feedback, the attitude of the spacecraft is such that the thrust vector is aligned with the desired reference vector r. Even though there is positive feedback in the path guidance minor loop, overall stability of the autopilot is not affected. Engine shut down occurs when a linear accelerometer indicates that the required ΔV magnitude is reached. At this time the motor burn switch disables the path guidance and enables the pitch and yaw gas systems. Attitude control is now returned to the gas system.

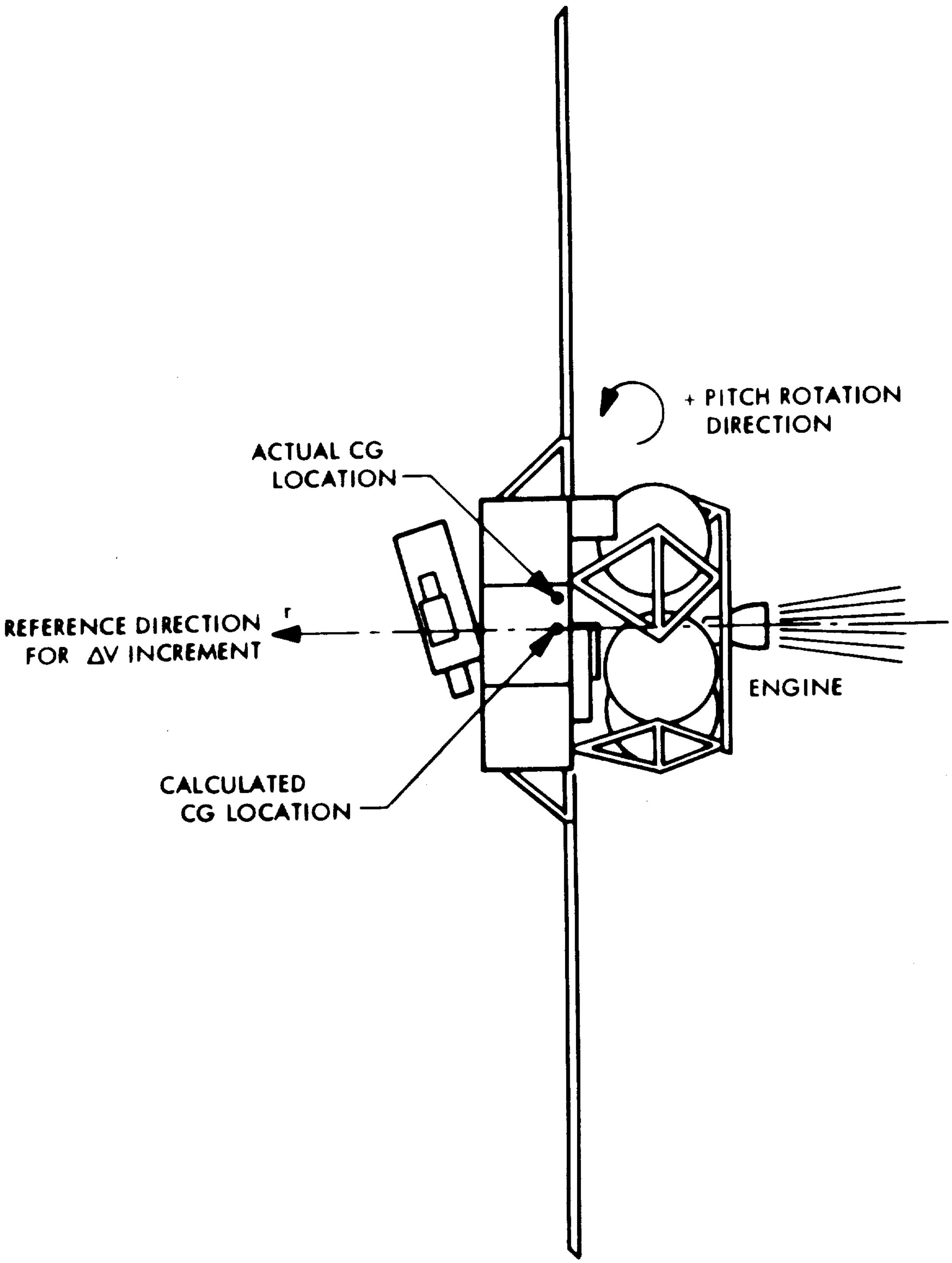

Figure 8I-13. Spacecraft Attitude at Motor Ignition

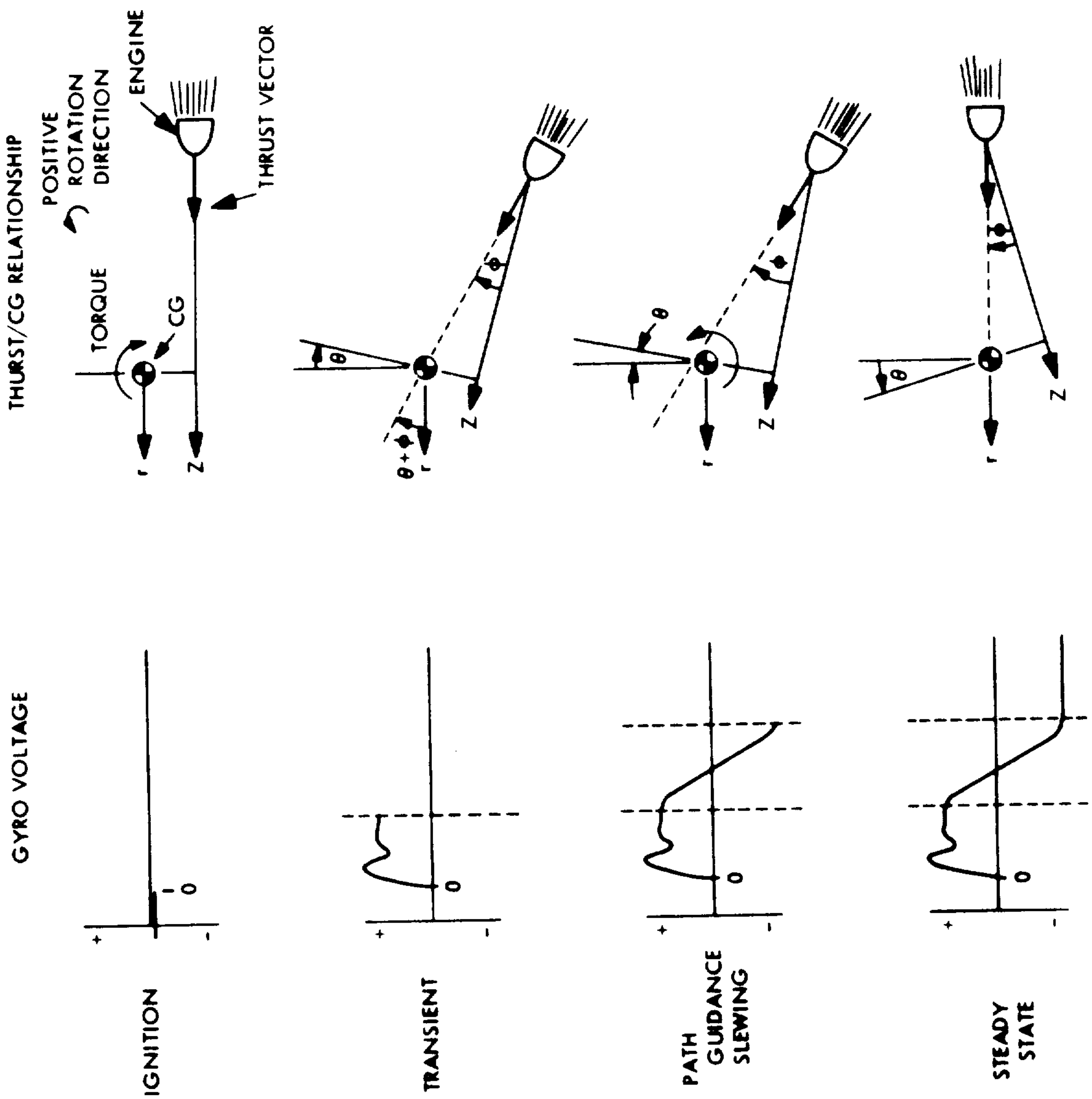

Figure 8I-14. Thrust CG Relationship During Motor Burn

(a) Path Guidance Autopilot Accuracy

When the spacecraft is at a steady state condition during the engine burn, the engine is aimed at the CG, and without path guidance, makes an angle ϕ with the z axis as shown in Figure 8I-15. Since this angle is commanded by the autopilot, it is a known source of thrust vector error. Path guidance corrects for this error by setting the steady state gain between gyro input and gimbal angle equal to -1. For stability, the gimbal must point the engine through the CG, and thus the attitude angle of the spacecraft must be the negative of the gimbal angle. In theory, then, the thrust vector will be exactly in the inertial reference direction except for the mechanical alignment errors of the engine. The mechanical alignment errors which cannot be corrected for are, α_T, the angular error between the actual thrust and the engine geometric thrust axis, and α_E, the engine angular alignment error.

Referring to the autopilot loop in Figure 8I-12, it is seen that the path guidance feedback modifies the steady state gain. There is an associated lag, T_P, which is used to prevent this feedback from degrading the transient performance of the system. The feedback, K_P, is set by the relation

$$K_P = 1 + K_F$$

and

$$K_F = K_G K_D K_A$$

where

K_G = gyro position scale factor

K_D = autopilot forward loop gain

K_A = gimbal servo gain

The steady state error is

$$K_F\theta + K_P\phi = \phi = \alpha_T + \alpha_E + \frac{\delta}{d_1}$$

where

d_1 = lever arm

δ = CG offset

If K_P were exactly its nominal value, the steady state error would be

$$\alpha_{ss} = -\alpha_T - \alpha_E$$

However, due to thermal variations in K_G and the gimbal servo linearity, and

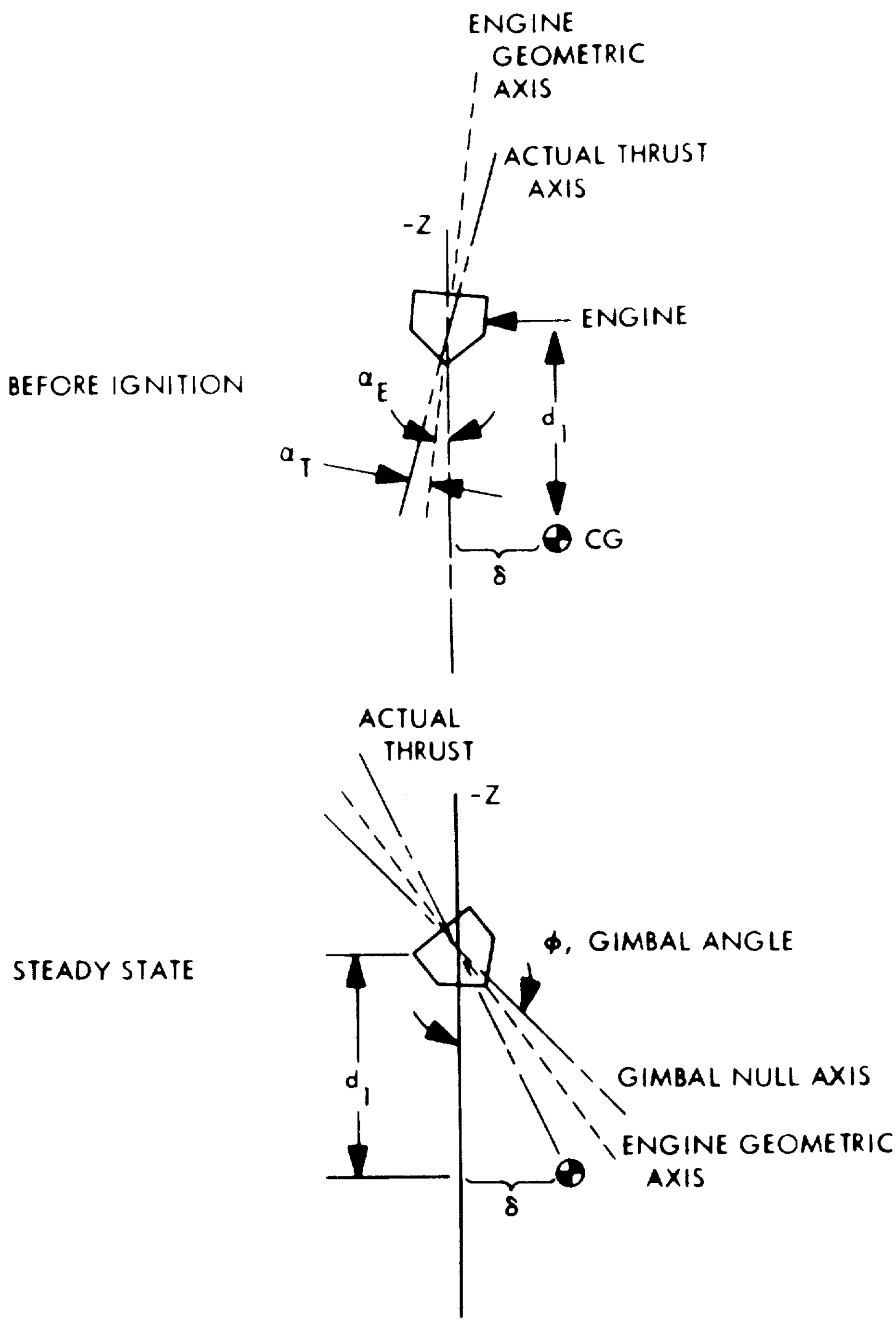

Figure 8I-15. Path Guidance Steady State Error

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the inability to perfectly match the gains such that the steady state gain between gyro input and gimbal angle is identically -1, an additional error is introduced. The steady state thrust vector error can be approximated by

$$\alpha_{ss} = -\alpha_T - \alpha_E + \frac{\Delta K_F}{K_F} \left(\frac{\delta}{d_1}\right)$$

where ΔK_F represents the aforementioned gain variations. $\frac{\Delta K_F}{K_F}$ is approximately 0.1 in the worst case and .06(3σ).

(b) Computer Simulation

To verify the autopilot analyses and assess the overall system performance, a six-degree-of-freedom computer simulation program was written for the IBM 7094 computer. The salient features of this program are listed in Table 8I-3.

The basic autopilot system described previously in this section is very similar to the MM71 system. The most significant difference that the autopilot must be able to cope with, is the much larger variation in spacecraft inertias and engine lever arm. The autopilot loop gain is given by

$$K_{AP} = \frac{K_G K_D K_A T \; d_1}{I}$$

Between the beginning of orbit insertion and the completion of capsule separation phase, the spacecraft pitch and yaw inertias undergo a 6.6 to 1 change while the engine lever arm undergoes a 2.33 to 1 change as shown in Table 8I-2. The net result is about a 3 to 1 change in autopilot loop gain. There are several criteria for establishing the correct loop gain. With regard to the Viking spacecraft, the most important criterion is establishing the gain such that a low frequency instability will not result. The low frequency instability is a secondary effect of path guidance on the system. The path guidance loop introduces a lower gain limit into the system stability criteria. In addition to this, normal autopilot operation will exhibit low frequency limit cycle behavior due to the inherent stiction thresholds in the gimbal actuators. The magnitude of these oscillations is determined by the threshold voltage necessary to overcome the

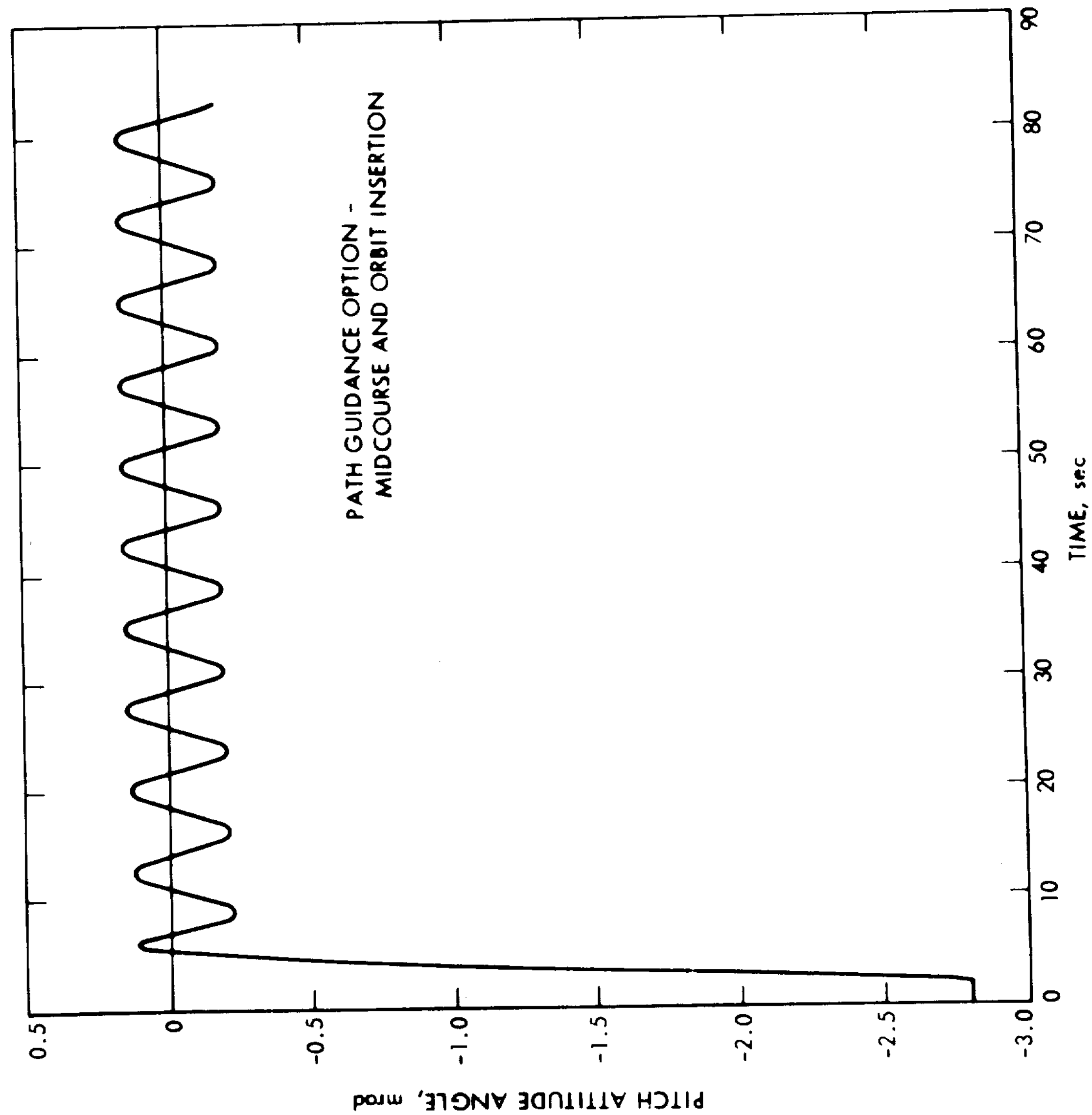
Table 8I-3. Autopilot Six Degree-of-Freedom Program Salient Features

SIX DEGREE-OF-FREEDOM DYNAMICS
CG OFFSET ERRORS
THRUST MISALIGNMENT ERRORS
TWO-AXIS ENGINE GIMBALLING
GYRO AND GIMBAL ACTUATOR DYNAMICS
COLD GAS ROLL CONTROL SYSTEM
GYRO DRIFT ERRORS
PATH GUIDANCE LOOP INCLUDED
SOLAR PANEL STRUCTURAL DYNAMICS MODELED
THRUST AND VELOCITY VECTOR POINTING ERRORS CALCULATED

3 4

gimbal stiction and the gimbal servo loop gains. The effect of path guidance is to raise this limit cycle frequency so that the period is approximately equal to T_P, the path guidance feedback time constant. Normally, this occurs without a change in amplitude. If the autopilot loop gain is maintained above the critical lower gain, the overall effect in system behavior is negligible.

Using the autopilot simulation program described above, the orbit insertion and midcourse maneuvers were simulated on the digital computer. The autopilot response and spacecraft response to autopilot control for the beginning of the orbit insertion burn and the midcourse maneuver burns are shown in Figures 8I-16 through 8I-21. Due to the large inertia and lever arm change that occurs during the orbit insertion burn, the autopilot loop gain was set for the conditions at the end of the burn subject to the constraint that a low gain instability would not result at the beginning of the burn. The orbit insertion burn will nominally be 45 minutes duration. Due to computer time limitations, the simulations were run long enough to reach steady state behavior. Figures 8I-16 through 8I-18 show spacecraft attitude during the initial 80 seconds of the orbit insertion burn. Engine ignition occurs at t = 2.0 seconds. The limit cycle behavior previously described is clearly shown. Its amplitude is approximately ±1.3 m rad. The initial negative going transient is due to solar panel structural disturbances, and the CG offset. The positive going transient and oscillations are caused by path guidance correction. The CG offset was chosen in the pitch direction, thereby creating the largest disturbance in the yaw axis. The roll position, shown in Figure 8I-18 remains at one side of the deadband. This is due to the roll gas system correcting for swirl torque. The yaw CG offset error was set at -.25 inches. This corresponds to an initial misalignment of .33 degrees. Figures 8I-19 and 8I-20 show the engine pitch and yaw gimbal positions during the burn. The limit cycle behavior caused by the non-linear stiction or threshold effects is readily seen in these figures. Figure 8I-21 shows the velocity vector error (deg) as a function of impulse (lb-sec). This curve is of use in determining velocity vector error as a function of ΔV. The initial transient shown is due to the CG offset, solar panel structural disturbance effects, and the inherent lag in the gimbal servo. If path guidance were not present, the velocity vector error which is proportional to the integral of thrust vector error, would remain at a value determined by the CG offset and lever arm, and autopilot steady state error. However,



8I-16. Pitch Attitude Angle for Path Guidance Option

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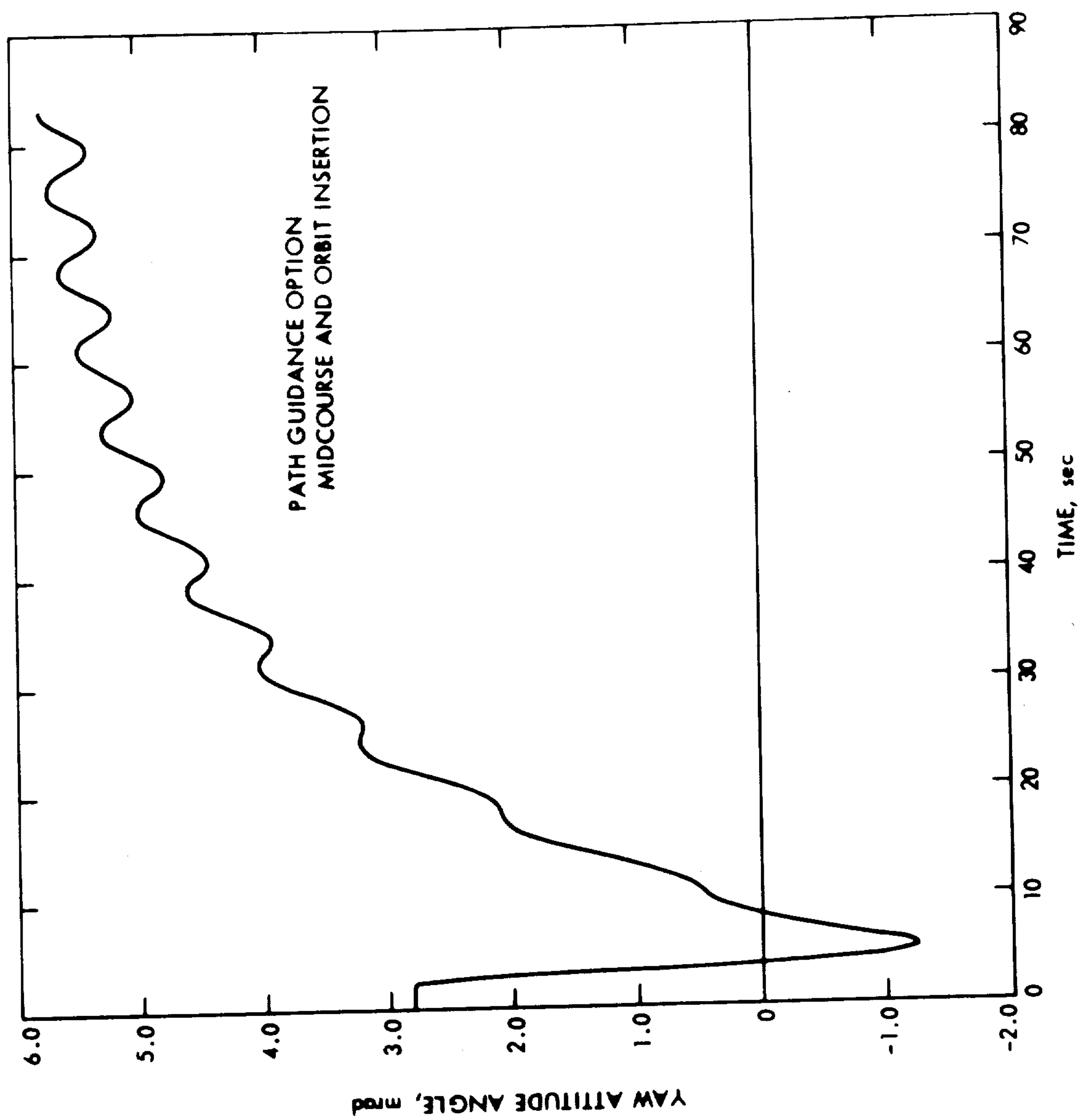

Figure 8I-17. Yaw Attitude Angle for Path Guidance Option

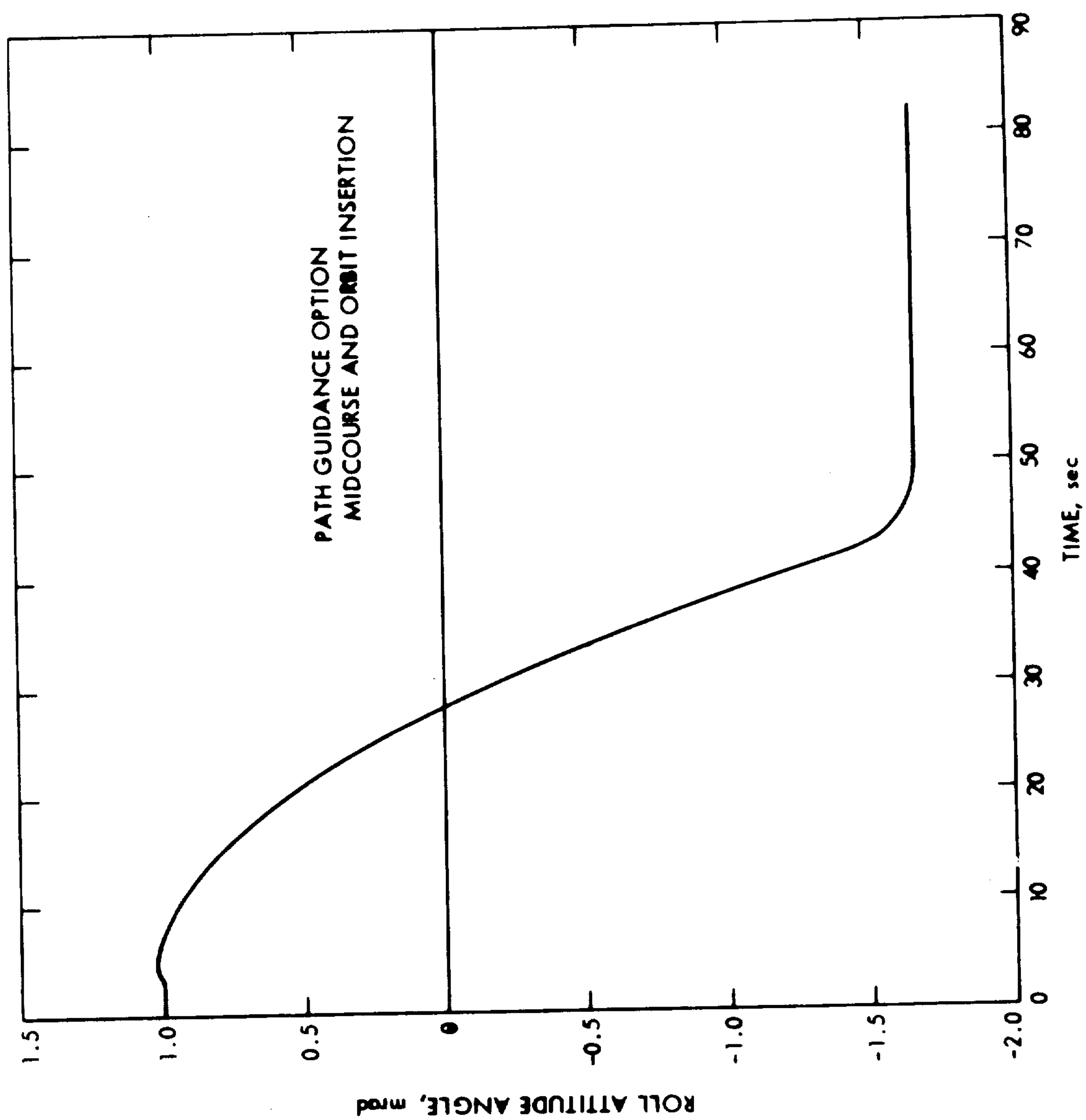

Figure 8I-18. Roll Attitude Angle for Path Guidance Option

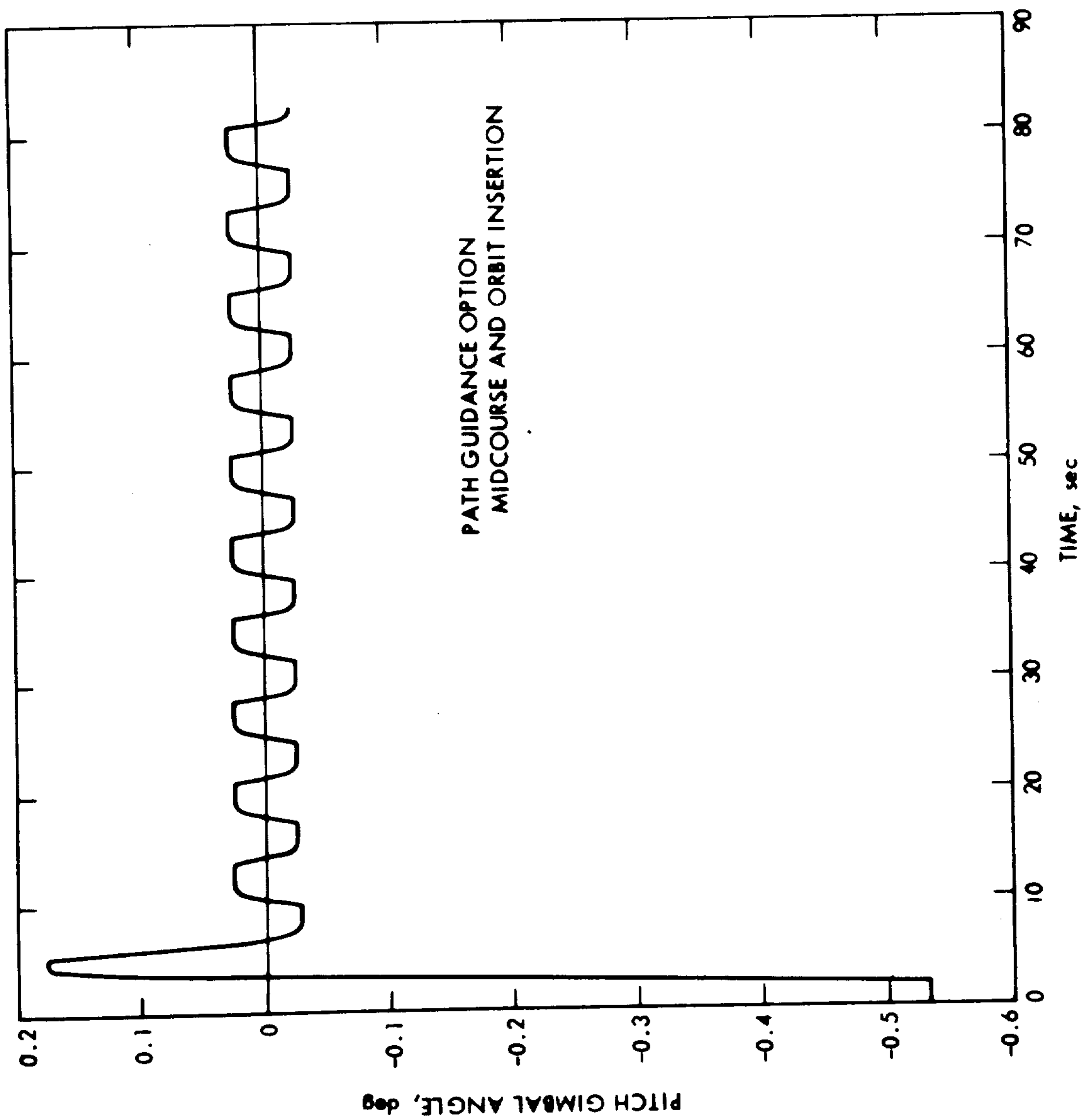

Figure 8I-19. Pitch Gimbal Position for Path Guidance Option

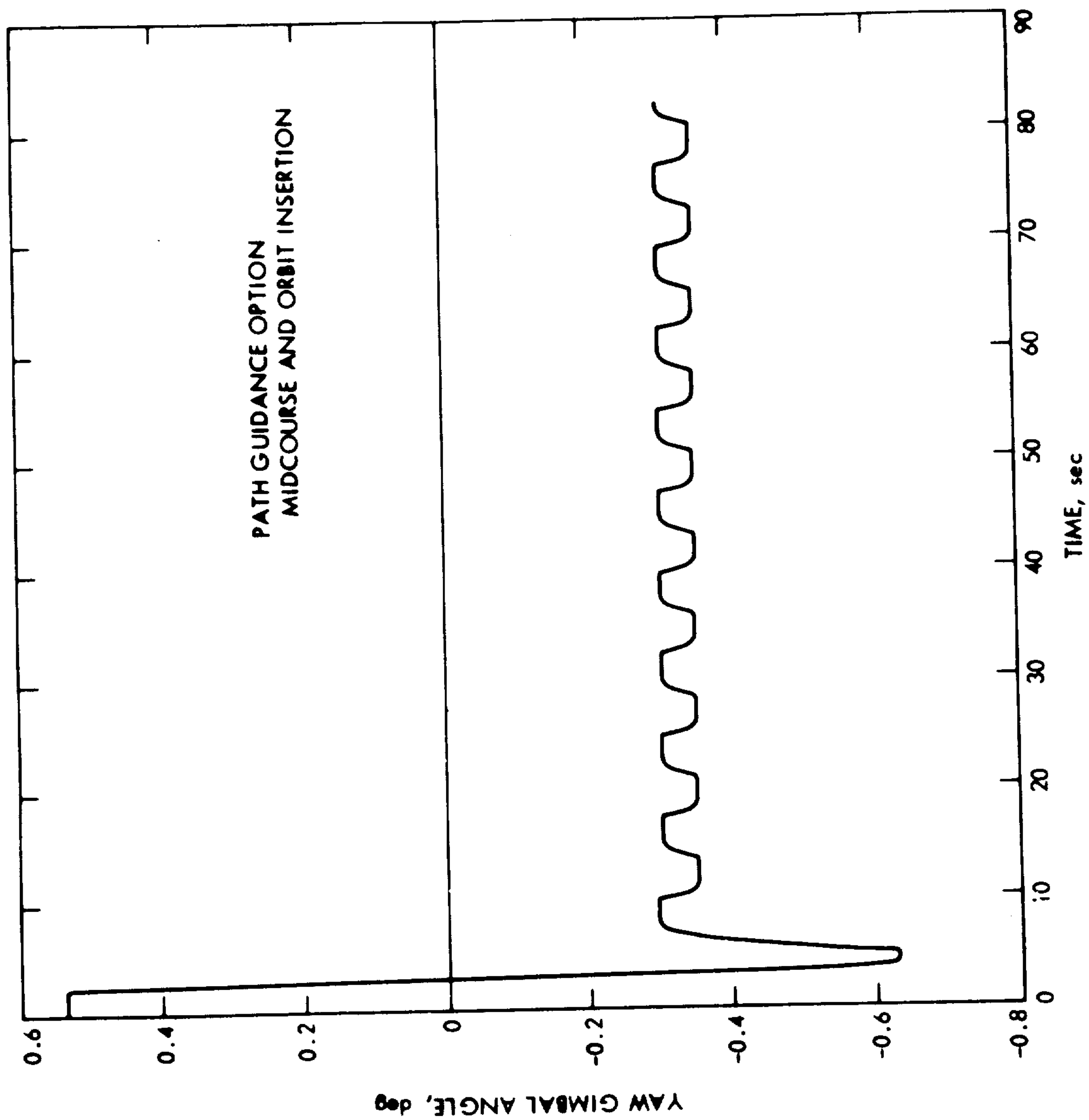

Figure 8I-20. Yaw Gimbal Position for Path Guidance Option

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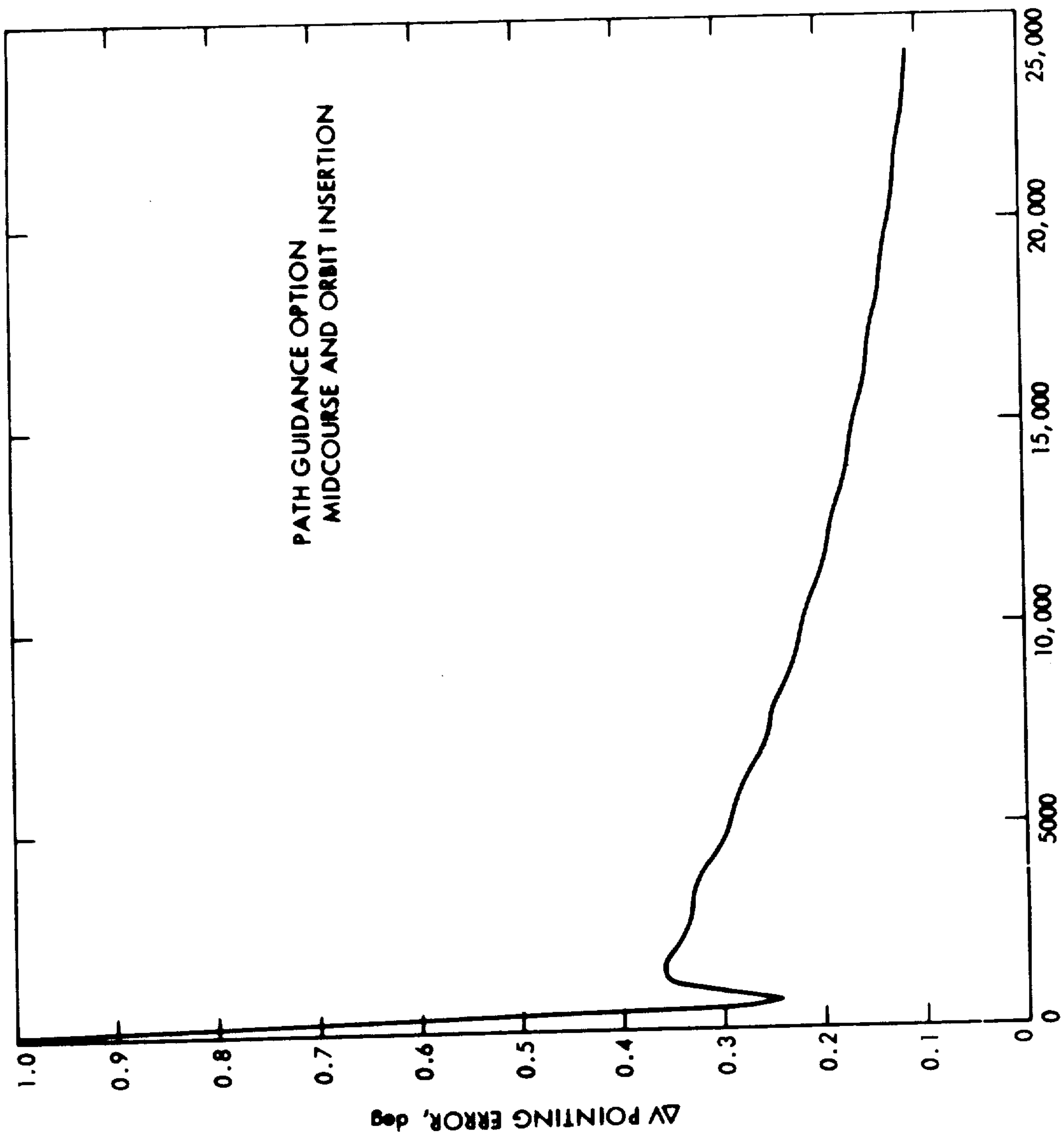

Figure 8I-21. ΔV Pointing Error vs. Impulse -- Path Guidance Option path guidance, in the manner described previously, reduces this error toward zero. For a 45 minute orbit insertion burn, this error would decay to its steady state value which is dependent on the mechanical alignment errors.

It should be noted that the transient response shown in the previous autopilot characteristics is slow due to the low loop gain condition at midcourse and the beginning of orbit insertion. The transient behavior for orbit trims occurring after orbit insertion and before capsule separation is considerably improved. This is because the loop gain has automatically increased due to the fuel depletion and resulting decrease in vehicle inertias. Loop gain at the end of orbit insertion burn is 7 db and -1.8 db at the beginning of the burn. However, if the loop gain is not changed at or before capsule separation, the even lower vehicle inertias after capsule separation will create an autopilot high gain instability. In the path guidance mechanization, the forward gain K_D, which sets the overall loop gain, and the path guidance feedback gain K_P, which establishes the proper steady state behavior, must be changed at capsule separation. This can easily be accomplished by a single relay switch closure that is initiated at capsule separation. These gains are changed such that the loop gain will remain at 7 db and the proper path guidance relation is maintained.

(2) High Gain Autopilot

The second autopilot design option is based on realizing through an appropriate compensation scheme, a high DC loop gain in order to minimize the cont rol system steady state error. This design is favorable if the CG migration is not severe, particularly during the long orbit insertion burn.

The basic autopilot is identical to the previous option except that the path guidance circuit will be removed, and the forward loop gain amplifier will be modified to provide both the high loop gain and high gain compensation. The pre-aim circuit discussed previously can also be used in this option to minimize the initial CG misalignment transient and also the steady state pointing error which is now only proportional to the CG offset error. A block diagram of the autopilot is shown in Figure 8I-22. The path guidance circuit has been removed and the forward gain amplifier modified for the appropriate compensation. The forward gain amplifier compensation is lag-lead.

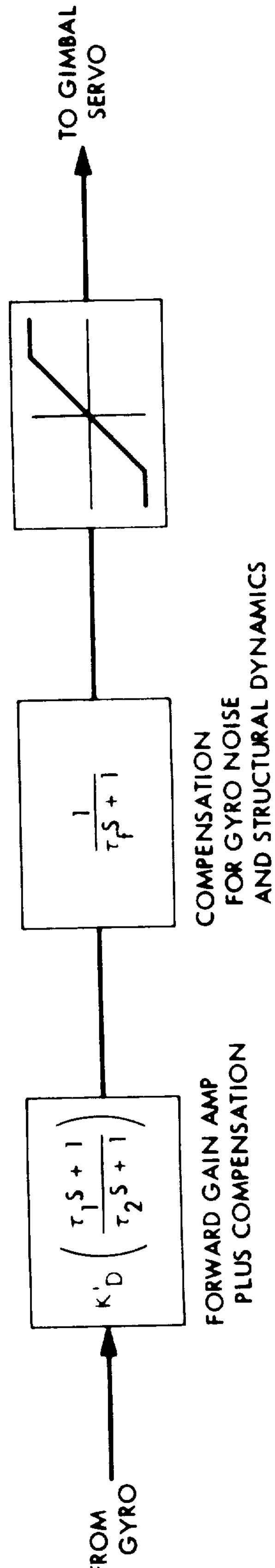

Figure 8I-22. High Gain Autopilot Block Diagram

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A linear analysis of the complete autopilot control system was made to determine the optimum values for τ_1 and τ_2. The results are shown in Figures 8I-23 through 8I-28. Figure 8I-23 shows a root locus diagram for the autopilot loop without the lag-lead compensation. This diagram shows that instability occurs for loop gains greater than 15.2 db. Figure 8I-24 shows the root locus for the compensated loop. Inspection of this figure and Figures 8I-25 and 8I-26, which shows the bode gain and phase plots, indicates that the system is conditionally stable. Instability occurs for gains lower than 5.0 (14 db) and gains greater than 115.4 (41.2 db). The loop gain is set at 17.8 (25 db). The values of τ_1 and τ_2 are

$$\tau_1 = .952 \text{ (zero at } s = -1.05)$$

$$\tau_2 = 20.0 \text{ (pole at } s = -0.05)$$

Inspection of the bode diagrams show that the gain margins are 10 db on the low gain side and 15 db on the high gain side. As in the previous option, K_D must be switched at capsule separation to maintain adequate stability margins. To verify the linear analysis, the actual nonlinear autopilot control system, including structural dynamics, was again simulated on the digital computer. The results are shown in Figures 8I-27 through 8I-32 for the midcourse and beginning of the orbit insertion burns. Inspection of these figures show that the autopilot and vehicle steady state behavior (except pointing error) is the same as in the path guidance option. The initial transient behavior is due to the gyro deadband initial conditions at ignition and an initial .25 inch CG offset.

The total steady state thrust vector pointing error (PTER) is given by

$$PTER = \frac{(\alpha_T + \alpha_E)}{K_F} + \frac{\delta}{d_1} + \frac{1}{K_F}\,\frac{\delta}{d_1}$$

$$\approx \frac{\delta}{d_1} = .33 \text{ deg}$$

where

K_F = gain between gyro input and gimbal angle

α_T = angular error between actual thrust and engine geometric axis

α_E = engine angular alignment error

δ = CG offset

d_1 = engine lever arm

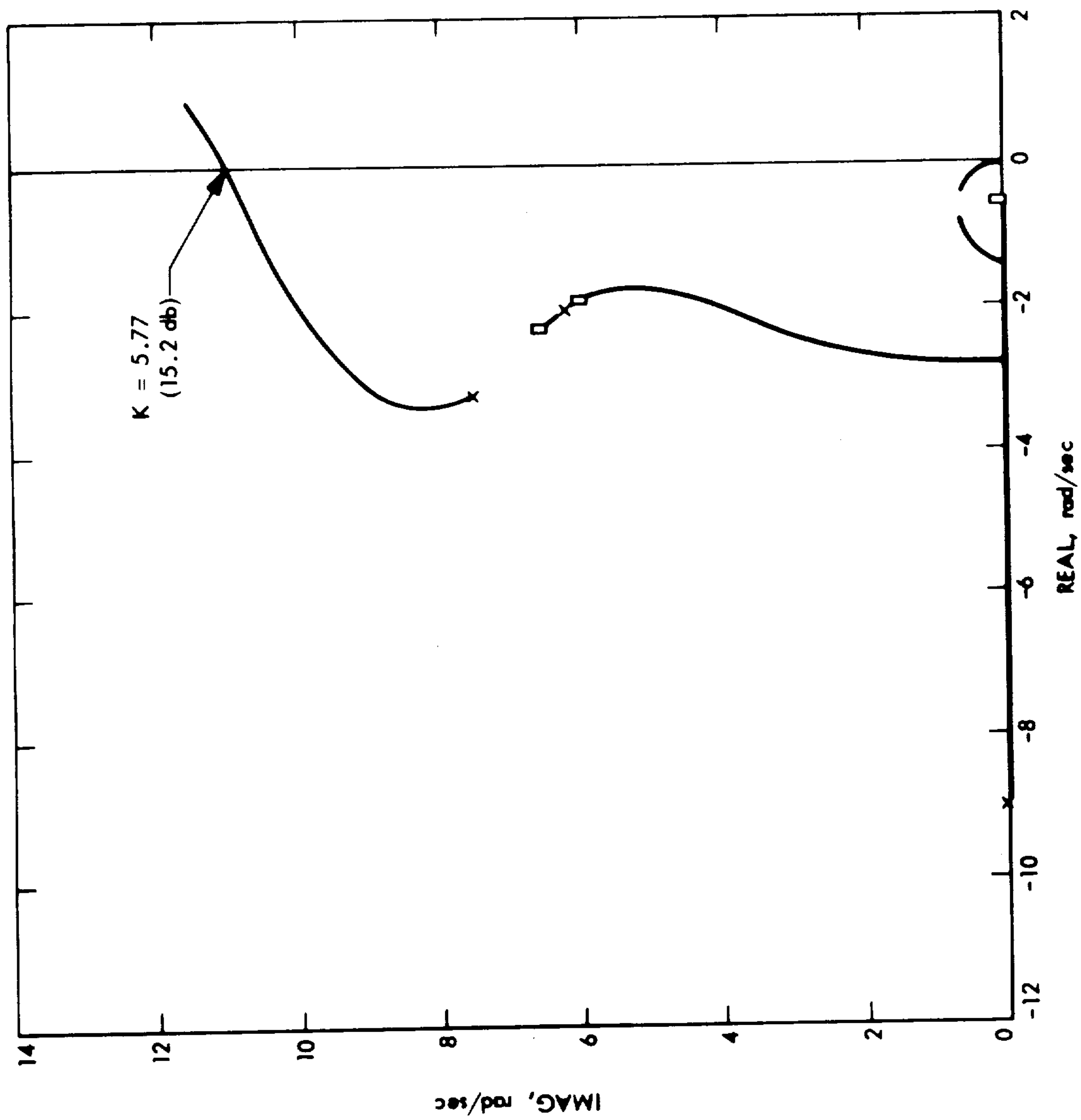

Figure 8I-23. Root Locus Uncompensated Autopilot

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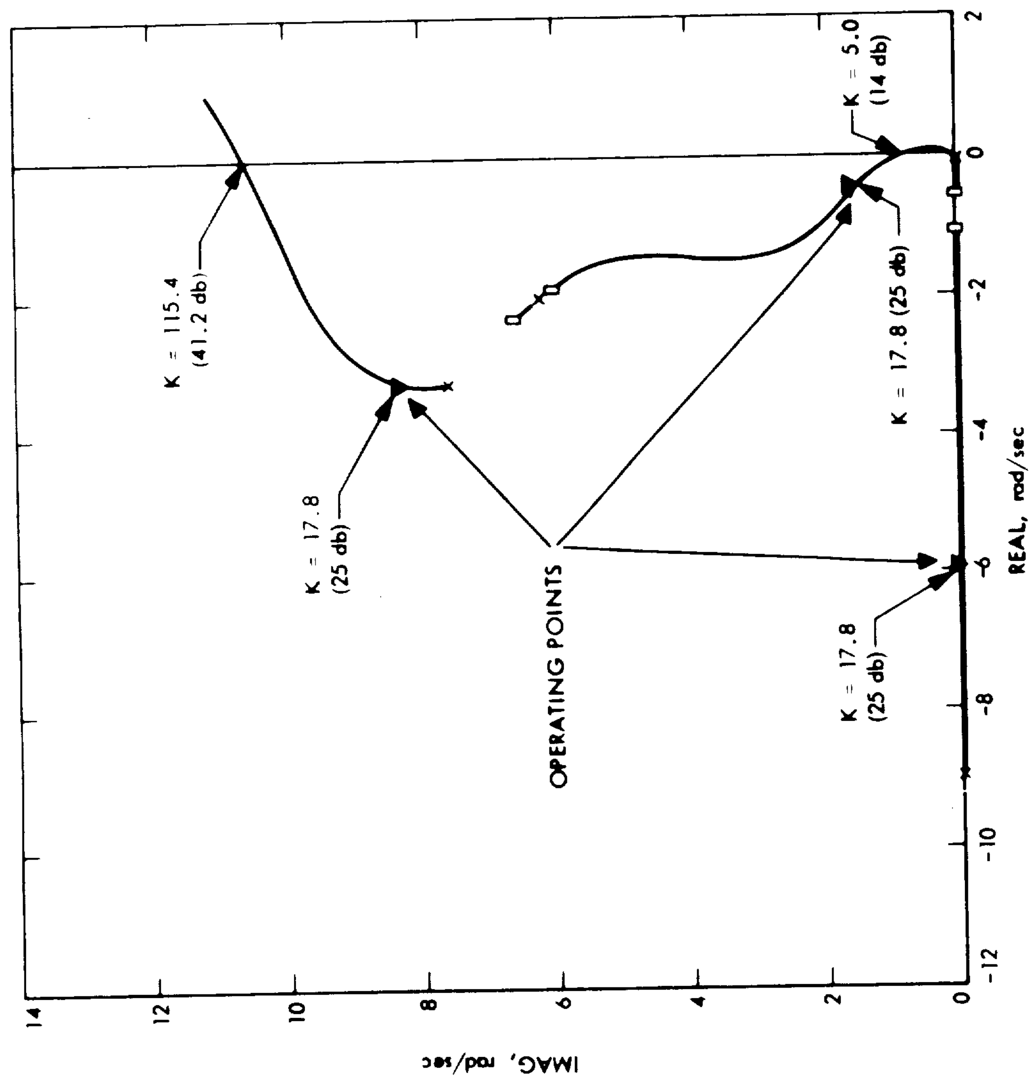

Figure 8I-24. Root Locus High Gain Option

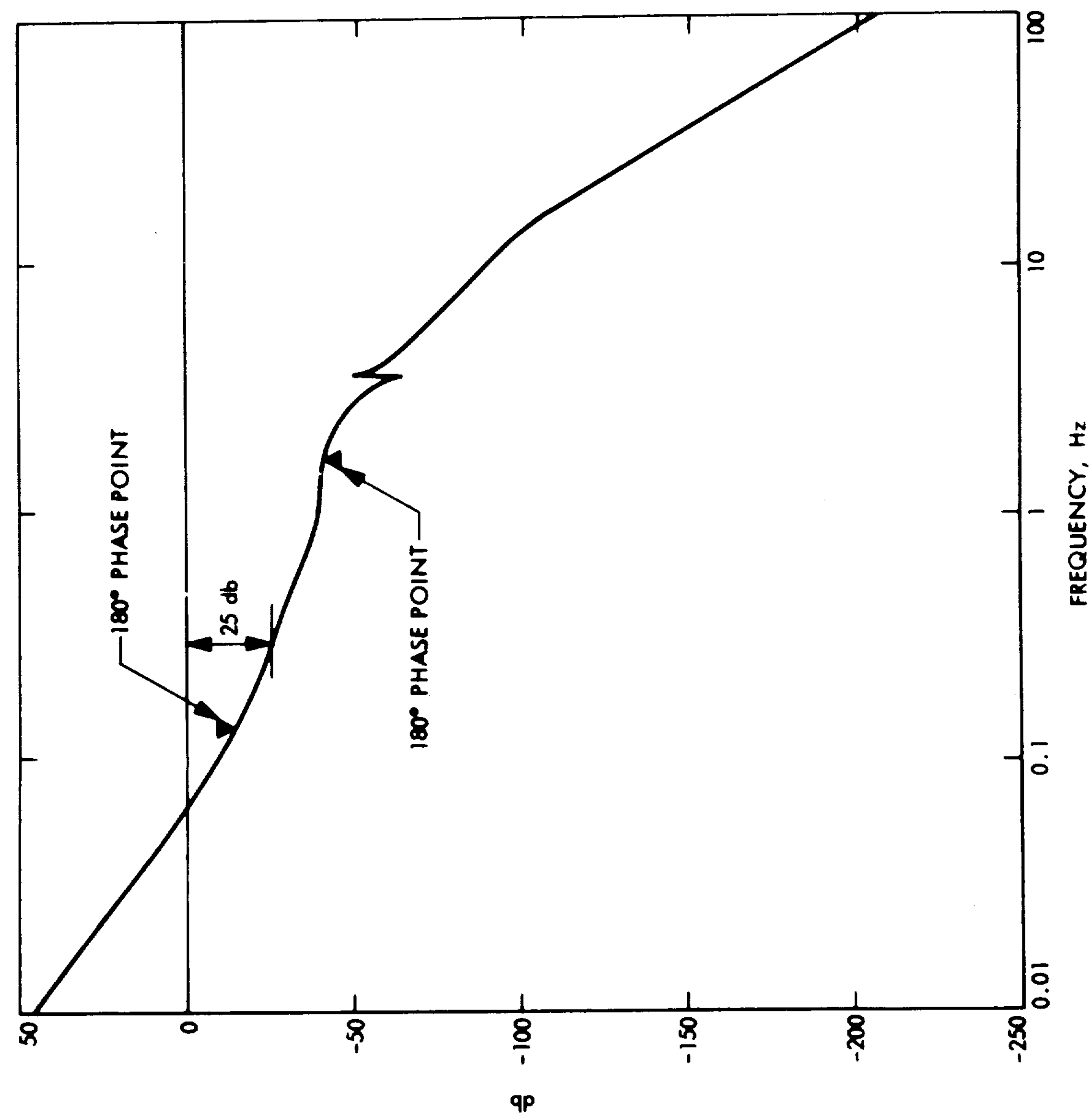

Figure 8I-25. Open Loop Gain vs. Frequency -- High Gain Option

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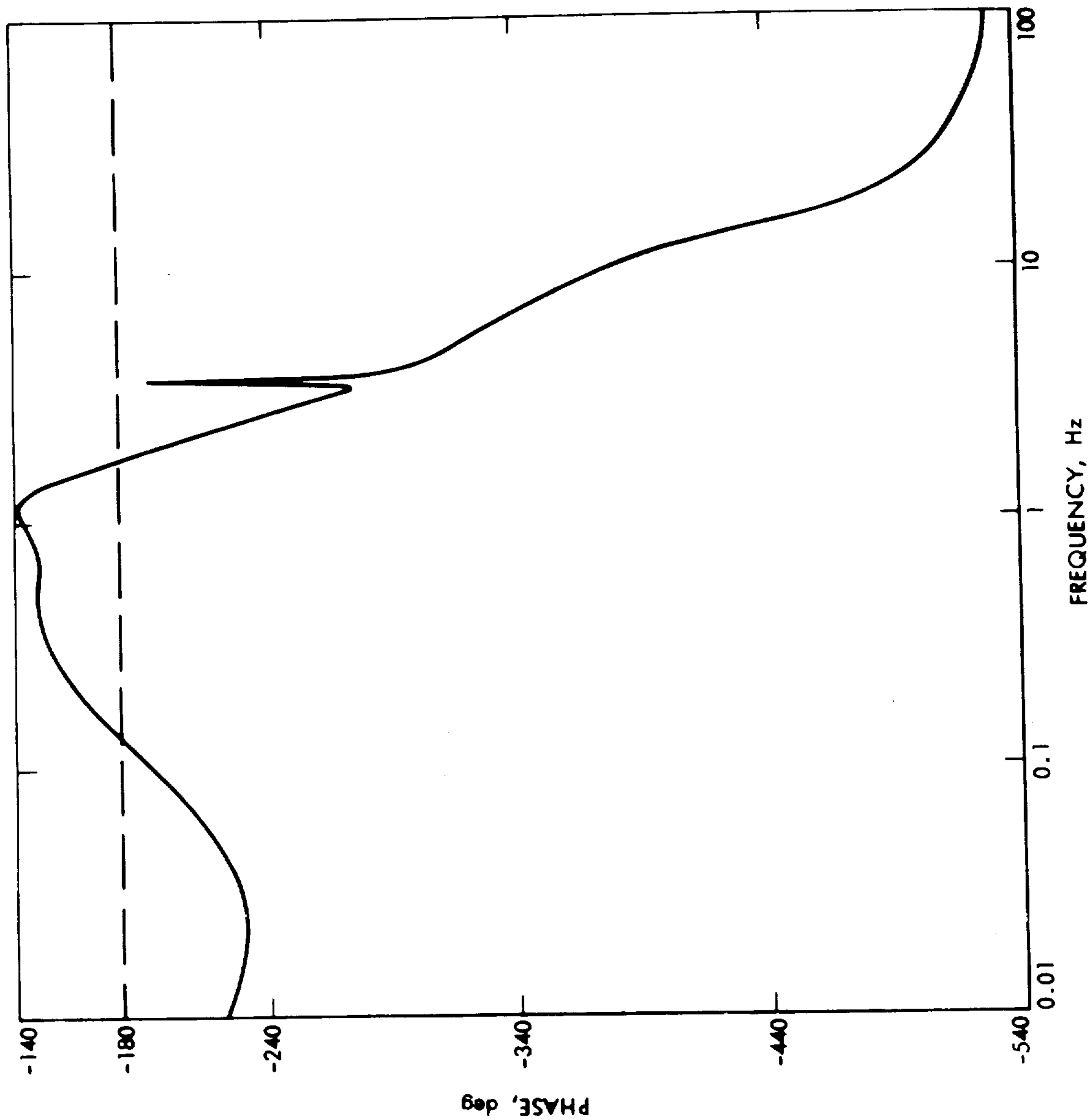

Figure 8I-26. Open Loop Phase vs. Frequency -- High Gain Option

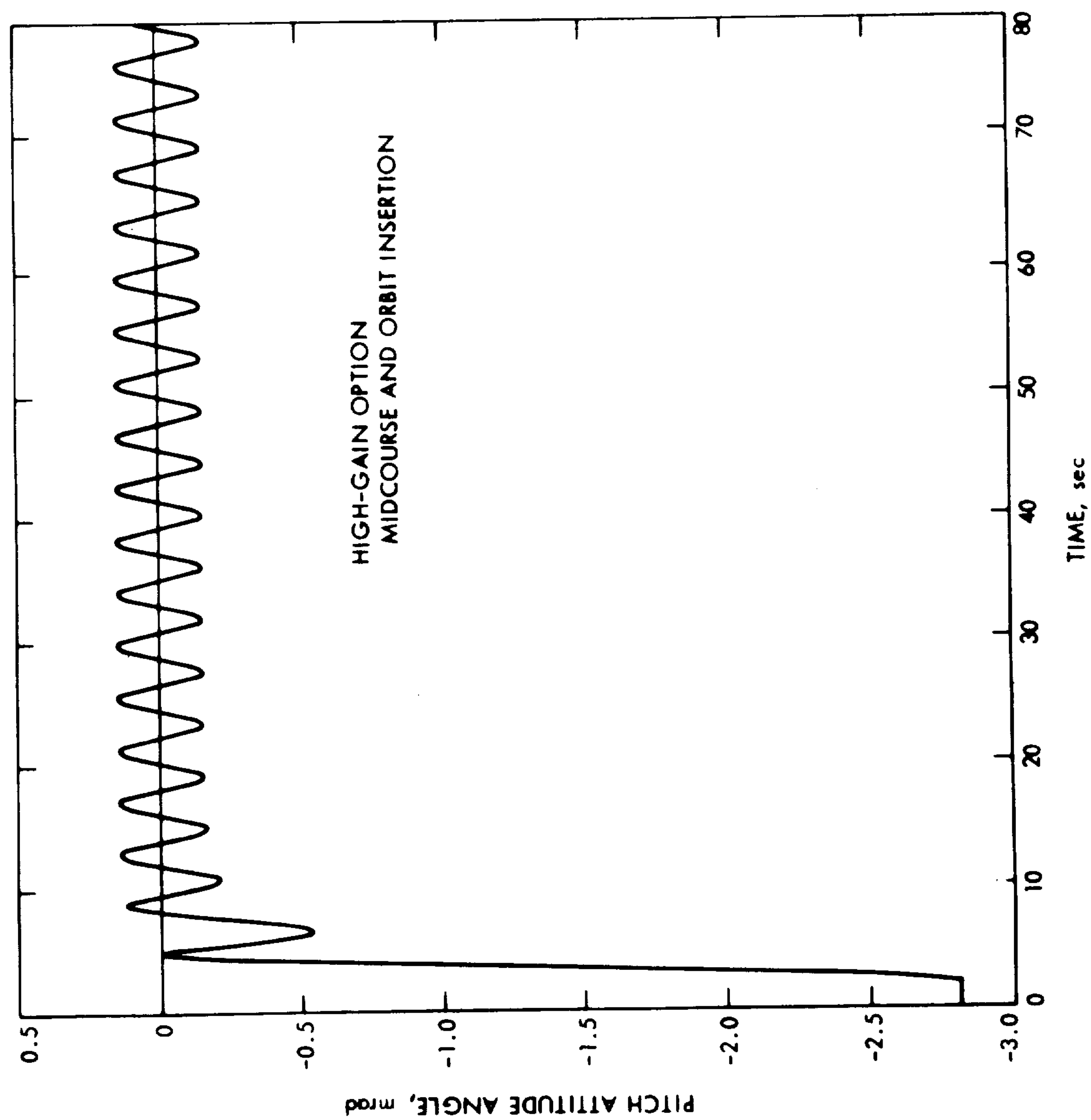

Figure 8I-27. Pitch Attitude Angle for High Gain Option

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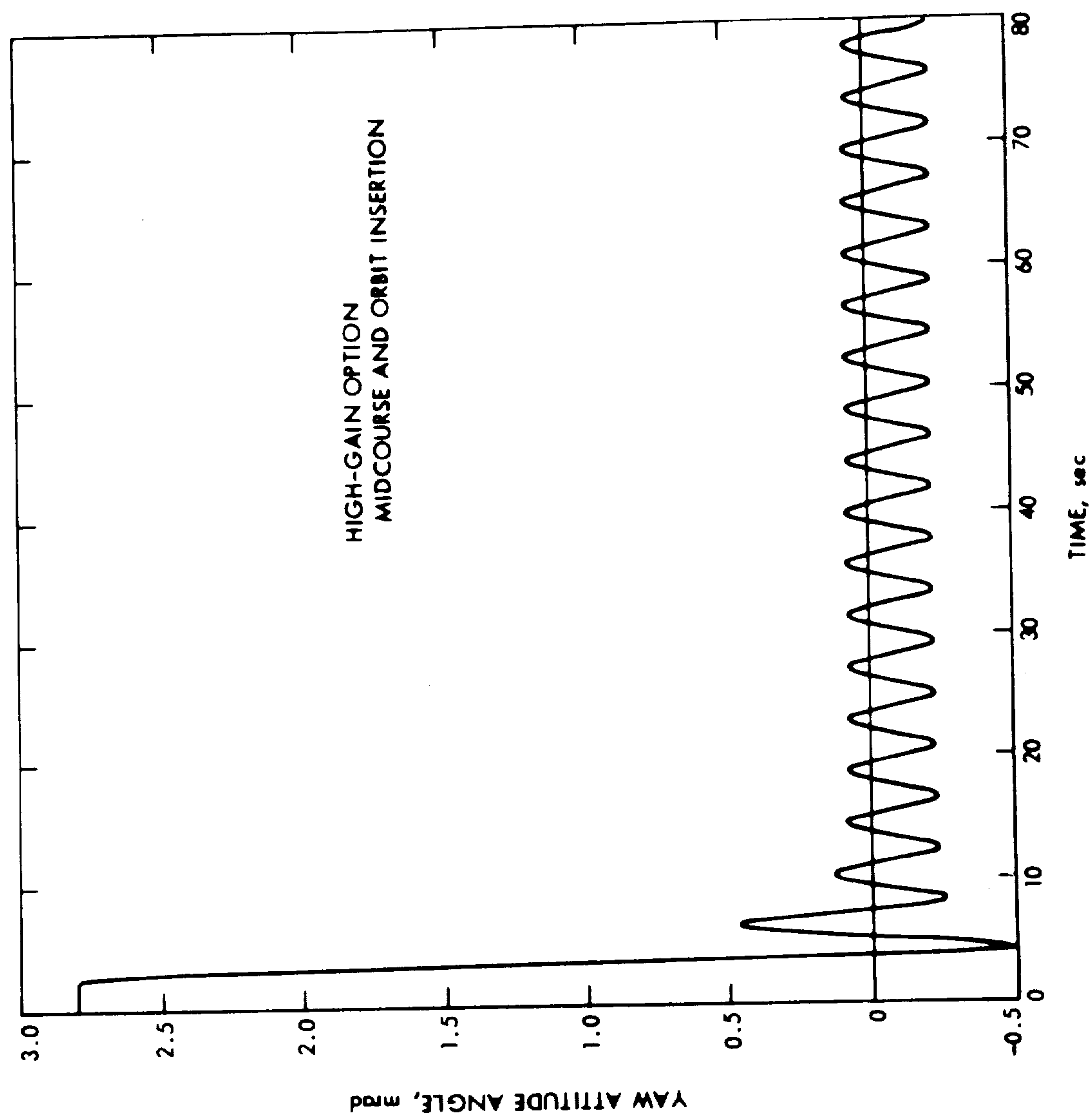

Figure 8I-28. Yaw Attitude Angle for High Gain Option

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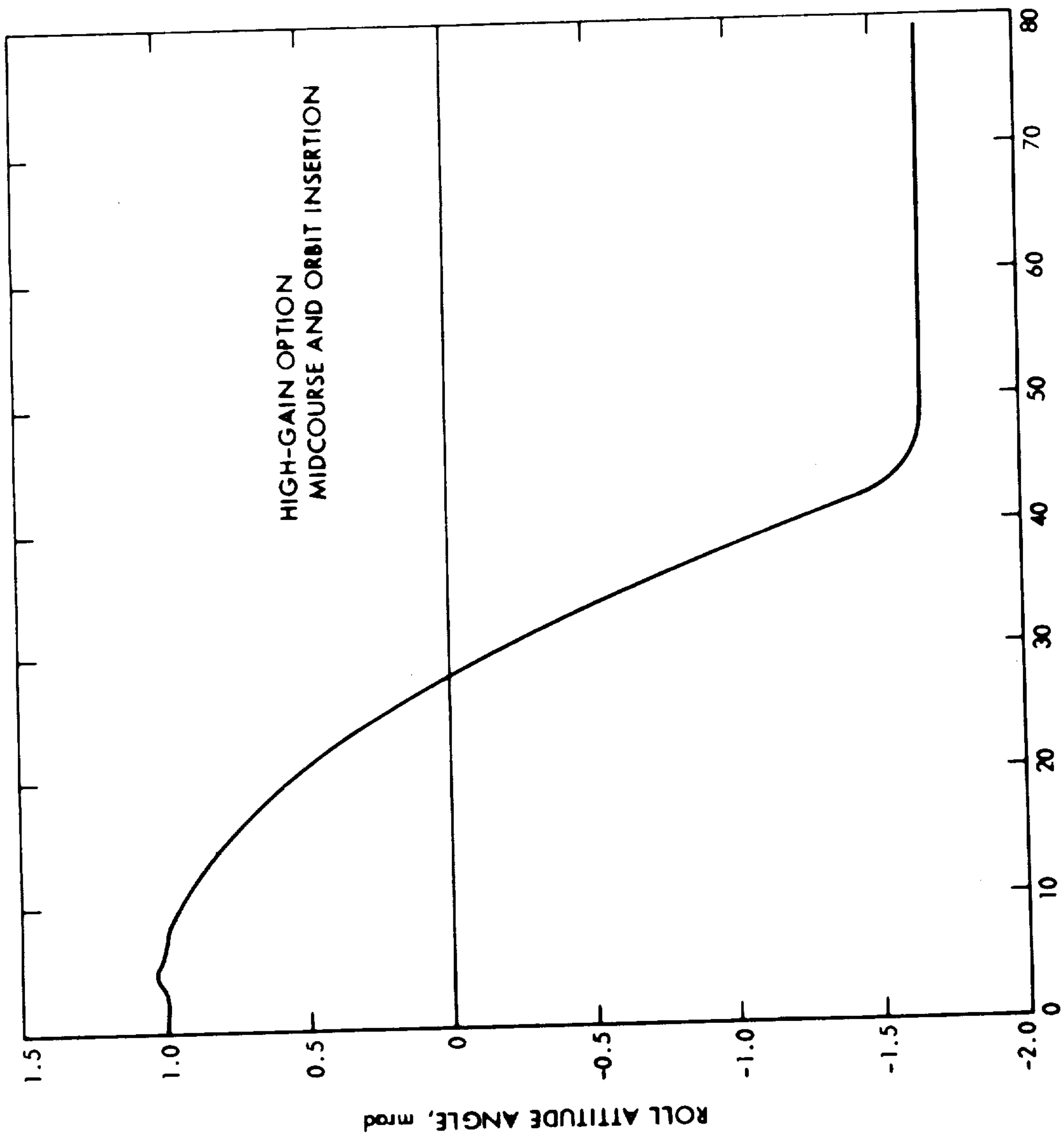

Figure 8I-29. Roll Attitude Angle for High Gain Option

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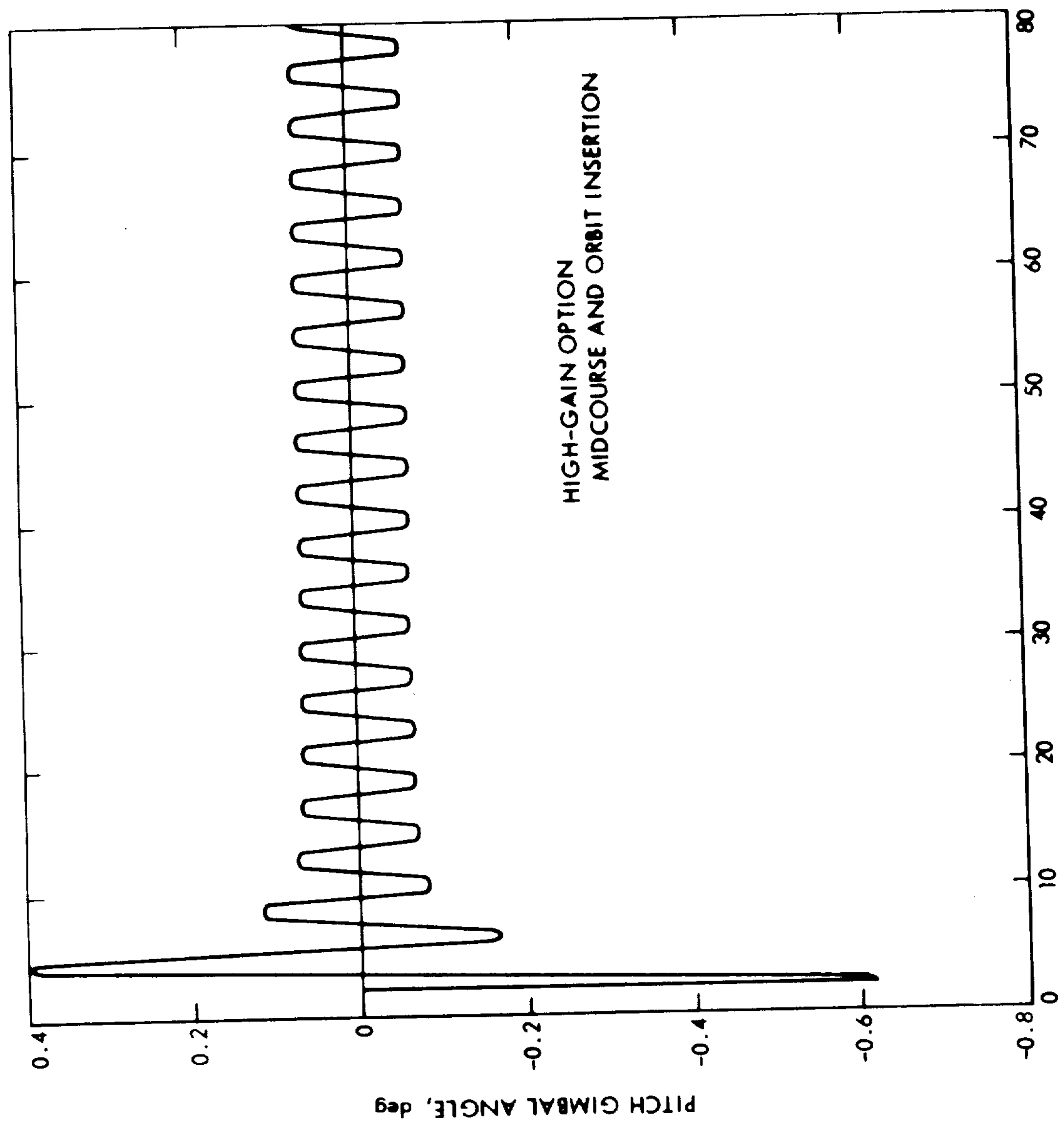

Figure 8I-30. Pitch Gimbal Position for High Gain Option

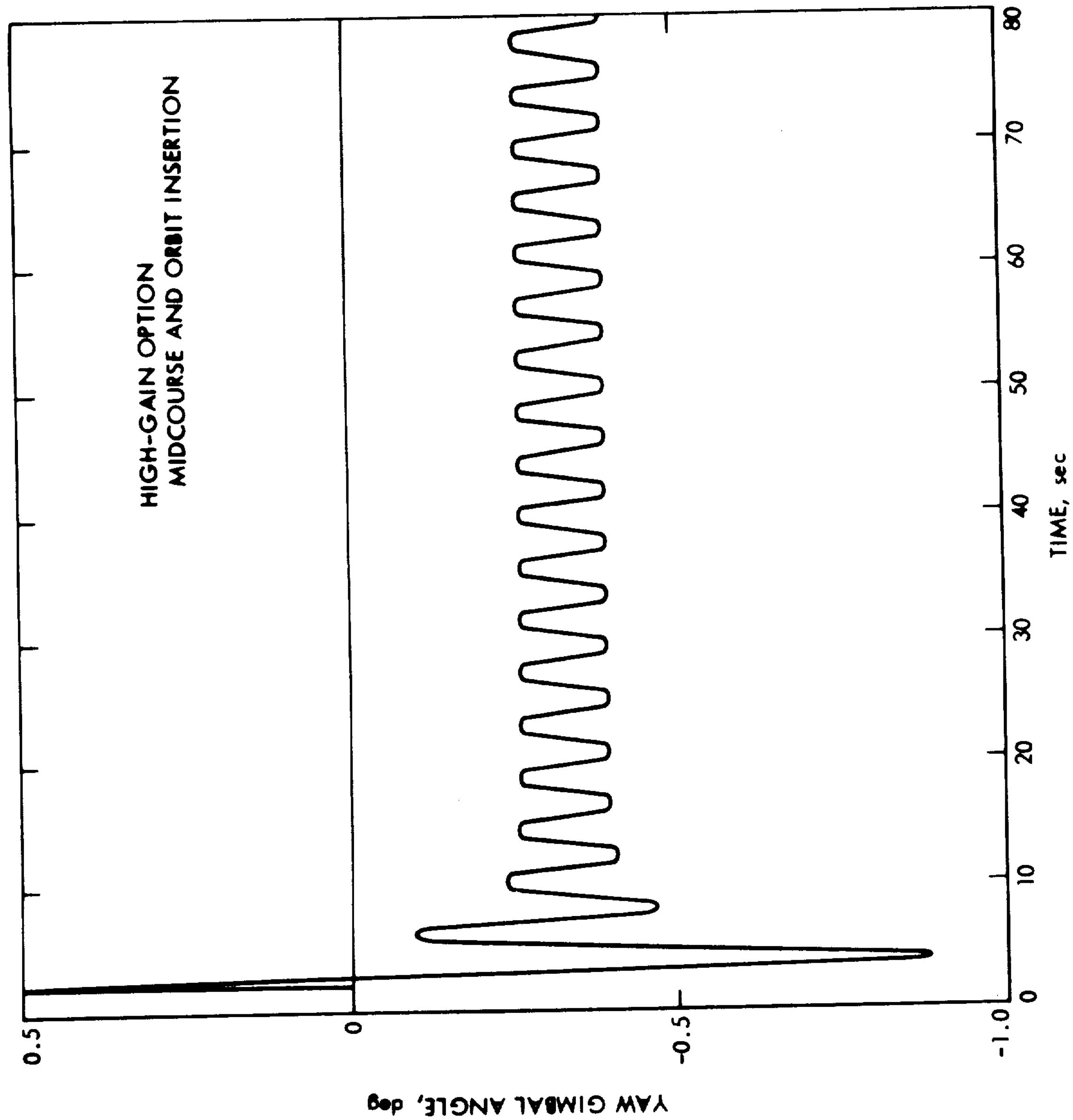

Figure 8I-31. Yaw Gimbal Position for High Gain Option

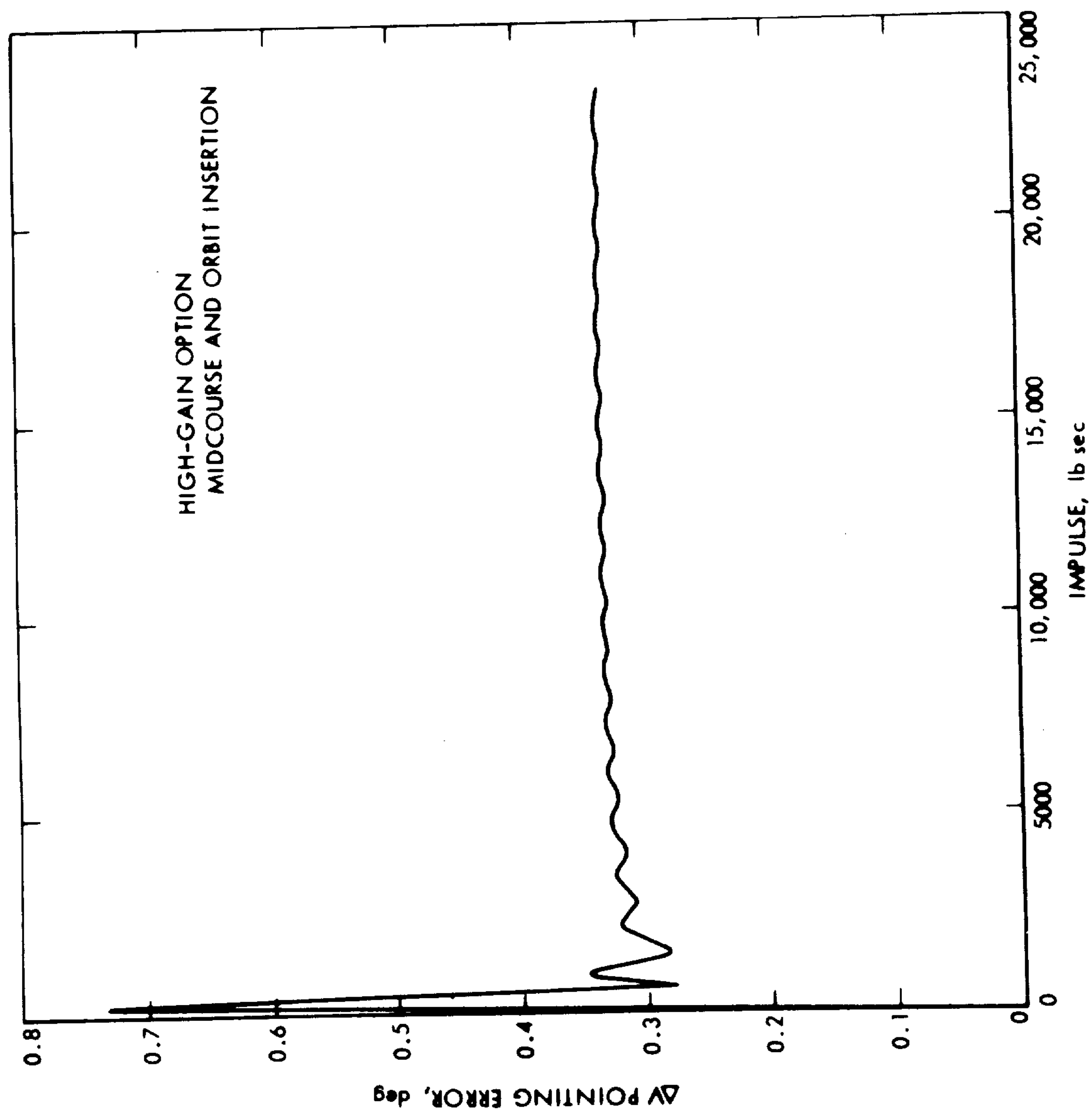

Figure 8I-32. ΔV Pointing Error vs. Impulse -- High Gain Option

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For the high gain autopilot option, the forward gain K_F is sufficiently large so that the steady state pointing error is only dependent on the initial CG alignment error. The value of .33 degrees corresponds to a CG offset error of .25 inch. For the initial midcourse, this error can certainly be held well within this value. For later motor burns, the use of gimbal pre-aim will insure the CG error does not exceed this value. This autopilot design, however, is not favorable if CG migration during the orbit insertion burn is severe.

c. Flight Control Subsystem Accuracy Analysis

During the mission the flight control subsystem must have the capability of performing midcourse, orbit insertion, lander orientation, and orbit trim maneuvers. The following is a brief look at the overall accuracy with which the flight control subsystem performs these maneuvers.

For the purposes of this analysis the first midcourse maneuver was assumed to consist of a roll turn, pitch turn and engine ignition. The expected, 3σ value of the velocity vector pointing error (per axis) for this maneuver is 20.3 milliradians.

For a second midcourse maneuver it may be necessary to perform a roll-pitch-roll commanded turn sequence (the second roll turn being used to point an antenna at the Earth). If this is the case, the expected 3σ value of the velocity vector pointing error (per axis) would be increased to 21.4 milliradians. During the midcourse maneuver the flight control subsystem also measures the spacecraft acceleration and turns the engine off after the appropriate velocity increment has been obtained. The expected 3σ value of the shut-off error for the midcourse maneuver is 0.35%.

For the orbit insertion maneuver a roll-pitch-roll commanded turn sequence followed by ignition of the engine was assumed. The second roll turn being used to point the maneuver antenna at the Earth. The expected 3σ value of the velocity vector pointing error (per axis) for this maneuver is 21.6 milliradians. Accurate accelerometer shut-off of the engine is more difficult for this maneuver due to the center of mass motion during the burns. The current estimate of the 3σ value of the shut-off error is 0.5%.

The lander orientation maneuver consists of a roll-turn followed by a pitch turn. The purpose of this maneuver is to orient the lander for the lander separation maneuver. The expected 3 σ value of the error (per axis) in orienting the orbiter is 16 milliradians. This represents only part of the velocity vector pointing error for the lander separation maneuver. Other errors such as orbiter-lander alignment errors, lander thermal deformation (after the bioshield is jettisoned), lander attitude control errors, and lander thrust vector control errors must be combined with this pointing error to obtain the total lander maneuver velocity vector pointing error.

The orbit trim maneuvers consist of a roll turn, pitch turn, roll turn, and engine ignition. As in the case of the orbit insertion maneuver the second roll turn is used to point the maneuver antenna at the Earth. Accurate control of the maneuver velocity vector is more difficult for trim maneuver because of the large center of mass motion during the orbit insertion maneuver and the center of mass changes when the lander is released. The expected 3σ value of the maneuver velocity vector pointing error (per axis) is 24 milliradians. The shut-off accuracy for the trim maneuvers is expected to be the same as that for midcourse maneuvers (.35%).

5. Weight and Power Summary

The flight control system weight and power requirements are summarized in Tables 8I-4 and 8I-5.

6. Additional Considerations

The different propulsion, thrust vector control and reaction control system configurations that were examined during the course of the study are discussed below. In addition, problem and areas requiring further study are listed.

a. Propulsion Configurations

The thrust vector control system provides thrust vector control during firing of the engine(s). In a single engine configuration three axis attitude control is accomplished by two axis gimballing of the engine with 3rd axis (roll) provided by the roll attitude control gas channel. This is identical

Table 8I-4. Flight Control Subsystem Weight Summary

FLIGHT CONTROL SUBSYSTEM BREAKDOWN	WEIGHTS (lbs)
N_2 Cold Gas	14.95
N_2 Tankage	23.91
Fixed Gas System Hardware	23.5
Gas System Subtotal	62.36
Attitude Control and Autopilot Electronics	8.5
Inertial Reference Unit (gyros, accelerometer, and electronics)	11.0
Sun Sensors, Sun Gate	.81
Canopus Tracker and Sun Shutter	9.2
Gimbal Actuators (2)	5.4
Antenna Control (1 degree of freedom)	10.0
Stray Light Sensor and Sun Occultation Sensor	5.0
Flight Control Subsystem Total	112.27

Table 8I-5. Flight Control Subsystem Power Profile (watts)

	AC 1	AC 2	GYRO 1	GYRO 2	AC 3	ANTENNA
Launch	18	11	10	9	25	3
Celestial Acquisition	30	0	10	9	0	3
Cruise	9	0	0	0	0	3
Midcourse Maneuver	28	11	10	9	25	3
Cruise	9	0	0	0	0	3
Orbit Insertion Maneuver	28	11	10	9	25	3
Sun Occultation	9	0	10	9	0	3
Orbit Trim Maneuver	28	11	10	9	25	3
Capsule Separation	9	0	10	9	0	
Orbit Cruise	9	0	0	0	0	3

AC 1	=	Attitude Control Power	2.4 KHz
AC 2	=	Autopilot Control	2.4 KHz
GYRO 1	=	Gyro Power	2.4 KHz
GYRO 2	=	Gyro Spin Motor Power	400 Hz 3Ø
AC 3	=	Autopilot Control Power	28 VDC
ANTENNA	=	Antenna Control Power	2.4 KHz

to the MM71 system. If, however, a three engine configuration is needed to meet the ΔV requirement, a more complex autopilot system will be required. Three axis control with a system employing three engines is accomplished by two axis gimballing the center engine for pitch-yaw control and single axis gimballing the outer two engines such that they provide a torque couple for roll control. The three engine configuration introduces several additional problem areas while aggravating many that are present with the single engine configuration. The development of a computer program for analyzing the three engine configuration is currently in progress and is near completion. The major disadvantages of the three engine configuration are listed below:

1) Increased sensitivity to CG offsets.
2) Increased dynamic range and faster response is required from the gimbal actuators. MM71 actuators cannot be used, thereby requiring a new actuator development.
3) Higher thrust level produces greater interaction between the control system and vehicle structure.
4) Variations in thrust level between the three engines introduces an additional disturbance.
5) Transient behavior is degraded from variation in start-up and shut-off times.
6) Lower shut-off accuracy.
7) More difficult mechanical alignment.
8) Additional autopilot channel and actuators are required.

From a thrust vector control point of view, the single engine configuration is definitely more favorable. However, one major problem in this approach is the gravity turn loss which results from the long burn duration that is required to meet the orbit insertion ΔV requirement. These losses can be reduced by performing a pitch turn during the orbit insertion burn.

b. Gravity Loss Considerations

A brief study has been made of the effect of thrust level and maneuver mechanization on the velocity magnitude required for the orbit insertion maneuver. Several assumptions were made for this study. They are listed in Table 8I-6. Based on these assumptions finite burn maneuvers were computed for different thrust levels and maneuver mechanizations.

The first mechanization considered was the fixed inertial maneuver where the thrust vector is pointed in a fixed inertial direction during the entire burn. The upper curve in Figure 8I-33 shows the resulting maneuver magnitude required for various thrust levels. The 300 lb thrust level corresponds to a single Mariner '71 engine while the 900 lb thrust level corresponds to three of these engines.

The second mechanization considered was an approximation of the gravity turn maneuver called a pitch-over maneuver (see Figure 8I-35)*. This maneuver consists of a roll-pitch-roll commanded turn sequence (the second roll turn is used to orient the spacecraft pitch axis) followed by a commanded turn about the pitch axis during the thrusting portion of the maneuver. The lower curve in Figure 8I-33 gives the maneuver magnitude for various thrust levels when this maneuver mechanization is used (The mechanization implications of a fixed rate pitch over are displayed in Figure 8I-34).

It can be seen from Table 8I-7 that smaller maneuver magnitudes are required when the pitch-over maneuver is used. The saving is greatest when one engine is employed (29 m/sec) and quite small (3 m/sec) for the three engines case. This mechanization provides a way of reducing the additional propellant required when one engine instead of three is used. However, the effect of errors in the pitch-over maneuver on the orbit trim budget has not been assessed at the present time.

c. Thrust Vector Control Configurations

In addition to establishing a thrust vector control configuration for the 3-engine propulsion system described in 6.a, additional analysis is

*During this study it was found that the amount of propellant required for a pitch-over maneuver was almost identical to that required for a gravity turn maneuver.

Table 8I-6. Assumptions for Gravity Loss Analysis

1. Approach trajectory and final orbit are conics.

2. Hyperbolic excess speed on the approach hyperbola is 2.85 km/sec.

3. The final orbit is an ellipse with a periapsis range of 4400 km and an apoapsis range of 36, 546.8 km.

4. The apsidal rotation is near zero. (The effect of apsidal rotation can be seen in Figure 8I-36).

5. The initial spacecraft weight at the start of the orbit insertion maneuver is 6465 lbs.

6. The specific impulse is 285 sec.

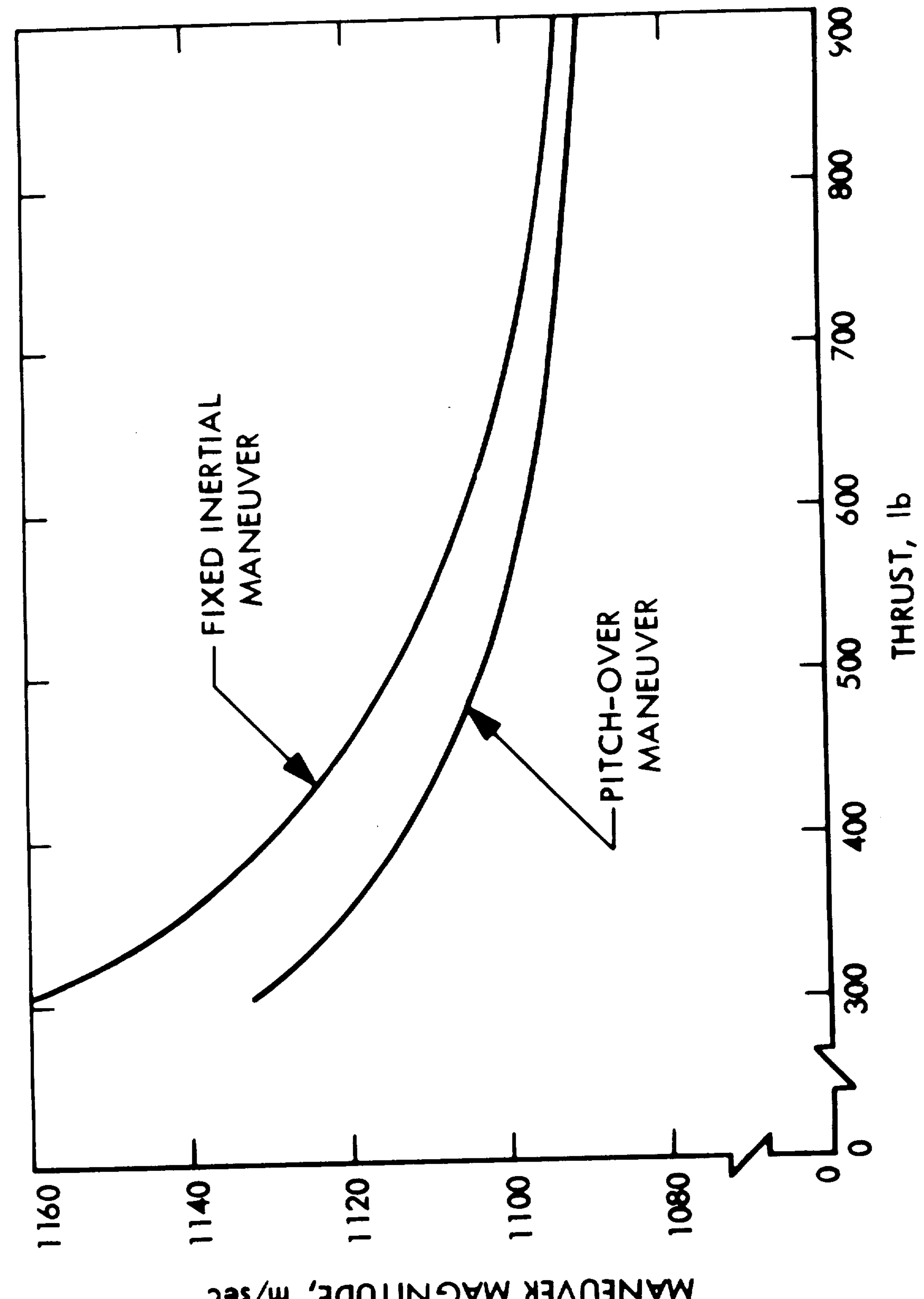

Figure 8I-33. Orbit Insertion Maneuver Magnitude vs. Thrust

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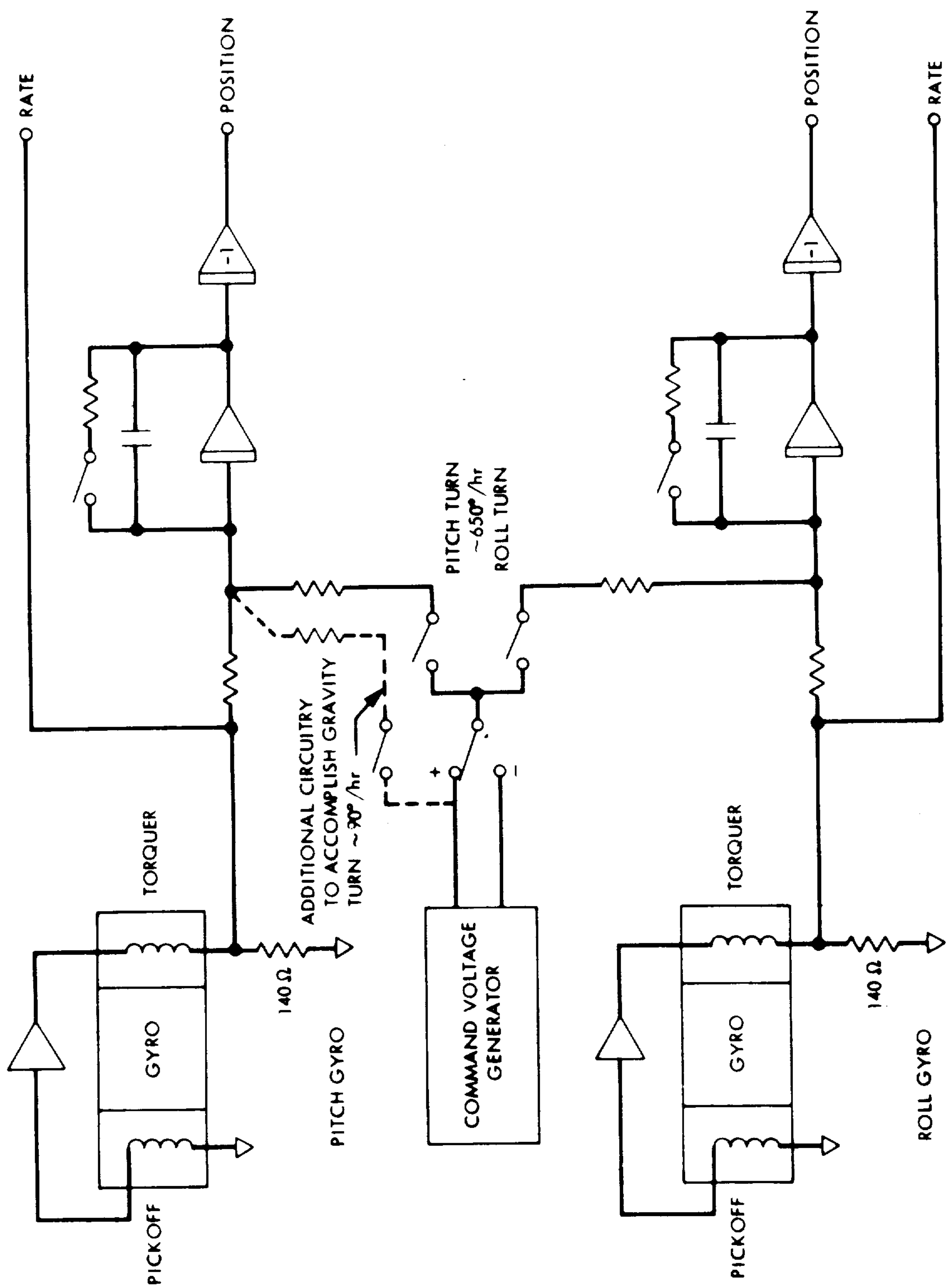

Figure 8I-34. **IRU Modification for Pitch-Over Turn**

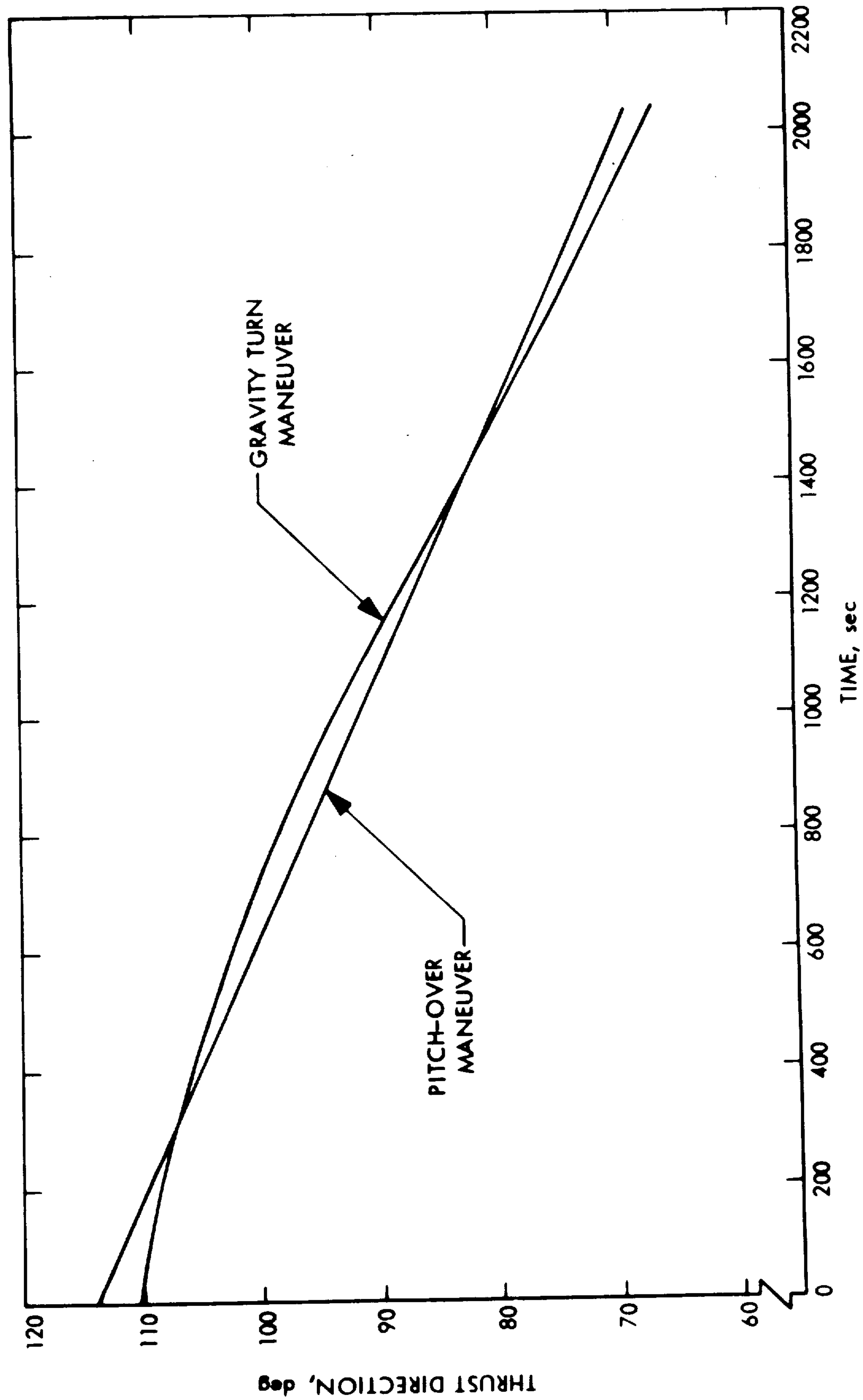

Figure 8I-35. Approximation of Gravity Turn by Pitch-Over Maneuver

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Table 8I-7. Orbit Insertion Maneuver Comparison - One and Three Engine Options

MANEUVER MAGNITUDES, m/sec

MANEUVER	ONE ENGINE	THREE ENGINES
FIXED INERTIAL	1161	1093
PITCH-OVER	1132	1090

PROPELLANT WEIGHT, lb

MANEUVER	ONE ENGINE	THREE ENGINES
FIXED INERTIAL	2198	2094
PITCH-OVER	2154	2089

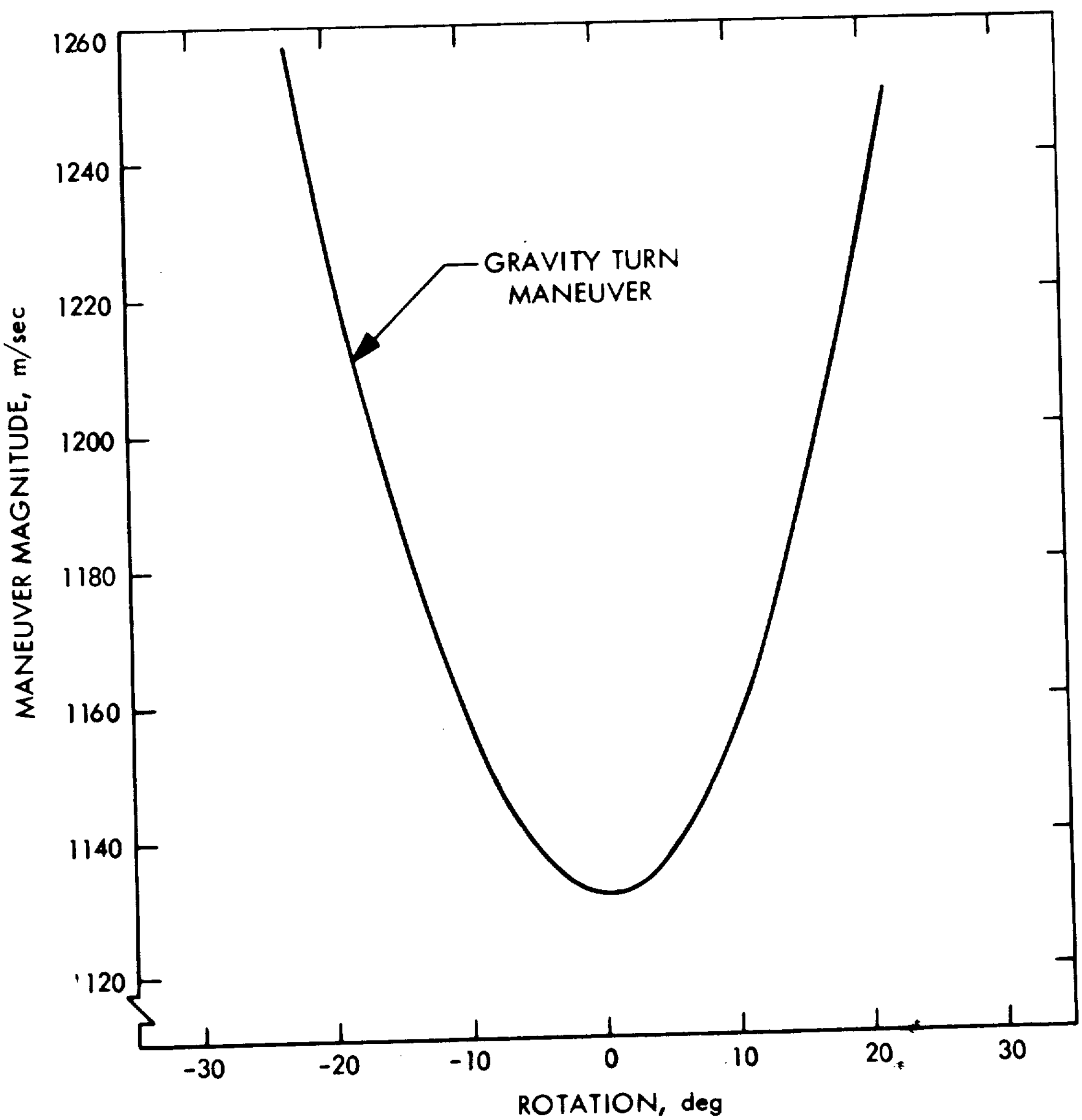

Figure 8I-36. Maneuver Magnitude vs. Apsidal Rotation required to select one of the two single engine thrust vector control configurations previously described in this report. The most important considerations with regard to final selection of a single engine thrust vector control system is CG behavior. The selection of a path guidance or high gain compensated autopilot will depend on the final determination of CG behavior. For example, the present orbiter design uses a symmetrical four propulsion tank configuration. It is anticipated that this configuration will create very little CG migration during the orbit insertion burn. If, however, a two tank configuration is selected, as on Mariner '71, CG migration may present the major source of pointing error.

Another area of concern is interaction between the vehicle structure and thrust vector control system. No problems exist with the present configuration. However, possibility exists that the scan platform may be moved substantially off the spacecraft z axis. If this is the case, excitation of the scan structure during motor burns will be more severe. A reexamination of this problem is required.

d. Reaction Control System

During all phases of the mission, except during motor burn phases, 3 axis control of the vehicle attitude is provided by the reaction control system. The system configuration is similar to that of the MM71 spacecraft. However, an increase in the gas storage requirement and gas jet thrust levels from the MM71 is necessary to meet the control requirements for the Viking spacecraft. These changes are summarized below:

1) Larger vehicle inertias.
2) Increased number of discrete events, particularly the number of commanded turns.
3) Increase gas jet thrust levels. As a result of the larger inertias, gas jet thrust levels must be increased to provide rapid acquisition of celestial references as required, proper limit cycle behavior, and proper vehicle response during commanded turns. In addition, the thrust levels must be sufficiently

high to remove spacecraft tip-off rates at capsule separation before large attitude errors result. This is because the spacecraft attitude is in inertial lock at capsule separation and the gyro integrators have limited angular storage capability.

The reaction control system that is presently being considered, utilizes a single gas jet thrust level per axis. There is a possibility that a saving in subsystem weight and an increase in overall efficiency may result if dual thrust levels are utilized. Dual thrust levels would compensate for the large vehicle inertia changes which occur between the transit and orbital phases of the mission. An examination of this system concept is currently in progress.

J. PYROTECHNIC SUBSYSTEM

1. General Description

The pyrotechnic subsystem for the Viking Orbiter is based upon proven techniques and design philosophy employed in previous Mariner spacecraft subsystems. Improvements in circuitry and components that enhance reliability will be incorporated. The design is essentially that of the Mariner 71 subsystem, modified only to the extent required to accommodate the increased propulsion events and added Propellant Migration Control (PMC) torque-motor valves.

2. Functions

The subsystem performs functions such as release of the Centaur/spacecraft separation device; firing of squibs associated with the propulsion subsystem explosive valves; control of the propulsion subsystem torque-motor propellant valve; release of the capsule adapter; release of the orbiter science scan platform and antenna(s).

Redundancy in critical functions is employed through the use of redundant commands, circuitry and devices.

The subsystem block diagram with functions and interfacing subsystems is shown in Figure 8J-1.

3. Subsystem Elements

Hardware elements included in the orbiter pyrotechnic subsystem are:

1) Pyrotechnic Switching Unit(PSU)--An electronic unit that contains a propulsion torque motor valve control section, an electroexplosive device firing section, and an instrumentation section. The unit employs a transistor switch to control power to the propulsion propellant valve torque-motor. Capacitor banks are used to store squib

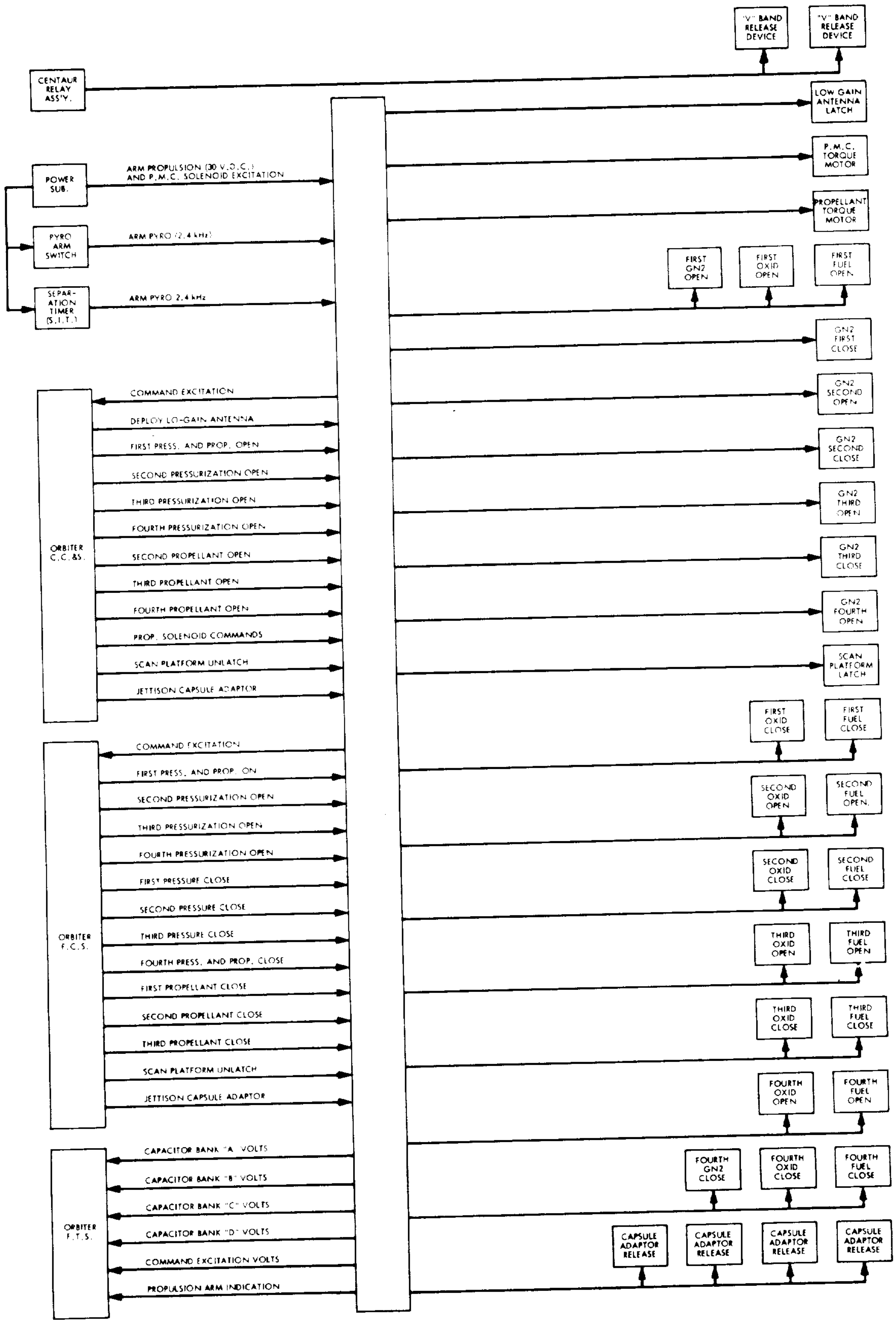

Figure 8J-1. Block Diagram of Viking Orbiter Pyrotechnic Subsystem firing energy; silicon controlled rectifiers switch the energy upon command to squib loads. Unijunction transistors are used to generate gate pulses for the SCR's. Advantages of such a circuit include lowered command current, assured triggering over a wide range of ambient temperatures, and reduction of false triggering due to noise. A schematic of a typical squib firing channel is shown in Figure 8J-2.

2) Electroexplosive Devices (EED) associated with the following functions:

a) Spacecraft/launch vehicle separation device.
b) Release of low gain antenna.
c) Release of the science scan platform.
d) Release of the capsule adapter.
e) Propulsion motor pressurization and propellant valves.

4. Subsystem Inputs

a. DC Power

DC power, regulated to $30\,^{+.6}_{-1.5}$ volts, 60 watts nominal, is required for excitation of propulsion PMC and propellant torque-motor valves. The power is made available to the PSU only when the Flight Command Subsystem supplies the proper command to the Power Subsystem. The power energizes the propulsion PMC torque motor immediately; the power to the propulsion propellant valve torque motor is interrupted by a transistor switch within the PSU. The switch is controlled by commands received from the Central Computer and Sequencer.

b. AC Power

AC power supplied to the subsystem consists of a 2.4 KHz square wave having an amplitude of 50 volts rms, +2, -3 percent. Average

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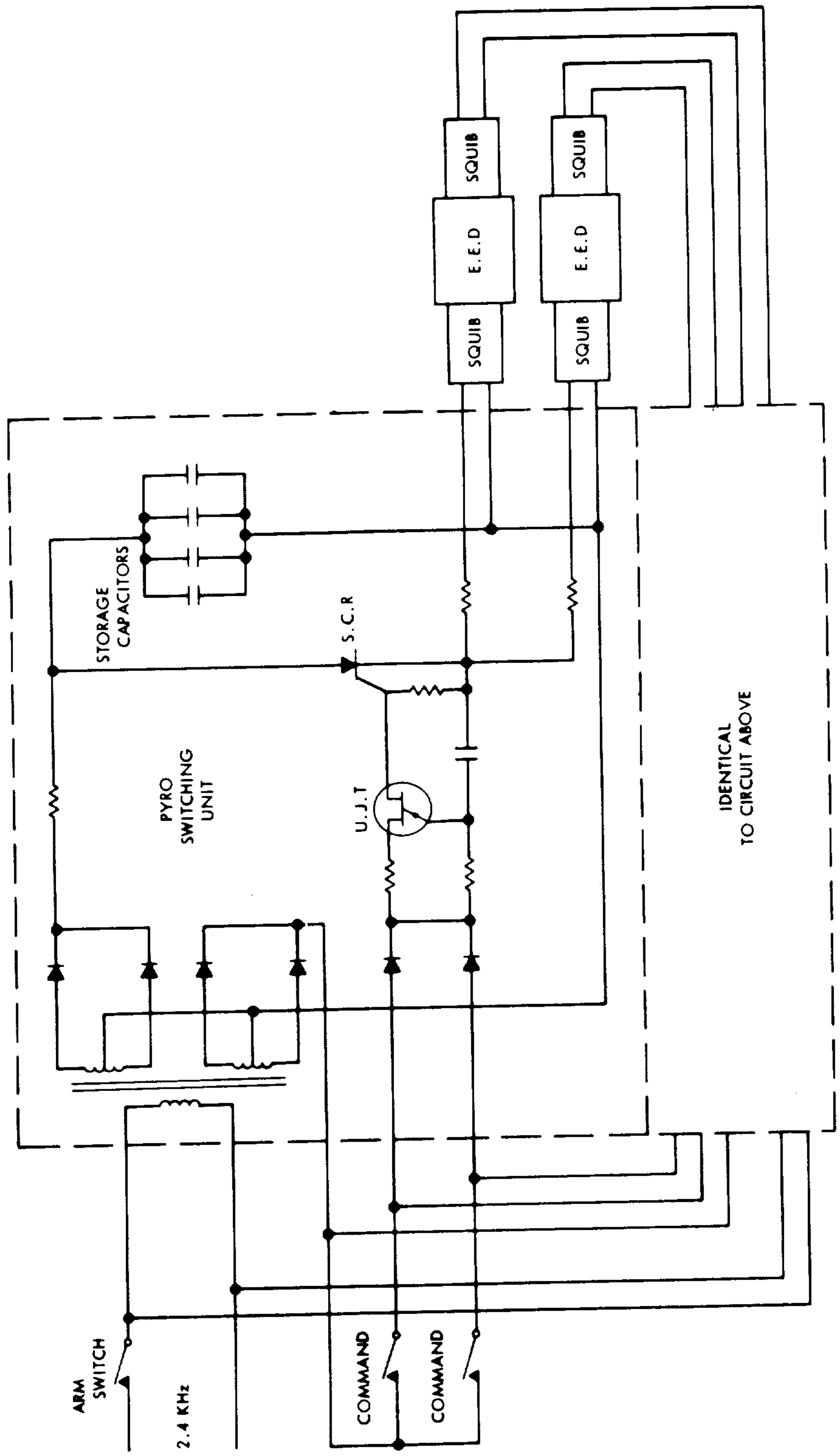

Figure 8J-2. Typical Squib Firing Circuit for Viking Orbiter PSU power demand is 1 watt. The peak power demand occurs with storage capacitor charging, and since capacitor charging current is limited, never exceeds 5 watts.

Multiple DC secondary voltages are provided through transformation and rectification for capacitor bank charging and pyrotechnic command excitation. The command excitation is distributed from the PSU to each orbiter subsystem required to provide pyrotechnic event commands.

Since the AC power to the subsystem is interrupted by the Separation Initiated Timer or Pyrotechnic Arming Switch, located on the Centaur/spacecraft separation adapter, the subsystem is not excited (armed) until separation has occurred.

c. Command Signal Inputs

PSU command signals consisting of a momentary closure of between 10.0 milliseconds and 10 seconds are supplied by:

Central Computer and Sequencer (CC&S)

Flight Command Subsystem (FCS)

The command to the propulsion propellant valve transistor switch is a circuit closure for the duration of each propulsion burn.

Charging of capacitor banks associated with squibs that are fired late in the flight sequence is controlled by series latching relays. Inhibit or enable commands received from the FCS set the relays to either position.

5. Subsystem Outputs

Outputs from the PCU are:

1) Command excitation power, not to exceed 35 volts DC, is distributed to each orbiter command source.
2) Propulsion torque motor propellant valve power of 30 watts, nominal, for the duration of each propulsion burn.
3) Electroexplosive device firing pulses having a minimum amplitude of 5 amperes for at least 50 milliseconds are

delivered to each squib bridgewire upon receipt of the proper command. Multiple bridgewires are fired simultaneously in some pyrotechnic events, in which case, current limiting resistors in series with each bridgewire limit the peak current to the bridgewire, provide isolation in the event individual squibs short, and provide a means for measurement of delivered squib current with ground Operational Support Equipment (OSE).

4) OSE instrumentation outputs consist of the aforementioned squib firing current outputs, capacitor bank voltages, command excitation voltage and 2.4-KHz and 30-volt DC arming indications. These outputs are made available at an easily accessible connector when the PSU is installed in the orbiter.

5) Telemetry outputs from the PSU to the Flight Telemetry Subsystem are provided to obtain in-flight diagnostic information. The data consists of the amplitude of the command excitation voltage and the voltage of each capacitor bank, a status indication of the 30 volt DC propulsion torque-motor excitation, and an event signal indicating the delivery of firing current to squib loads at each pyrotechnic firing.

6. Electroexplosive Devices

Characteristics of the electroexplosive devices used are:

1) Devices using squibs have an electrical connector, the connector being an integral part of the squib body.
2) Squibs have redundant bridgewires.
3) Squibs are hot-wire gas generating types that have a 1-ampere, 1-watt, no-fire characteristic. The squibs will be capable of withstanding a static discharge of 25 kilovolts from a 500-picofarad source.

4) Devices are non-detonating, non-venting, non-rupturing, and provide positive gas containment.

7. Subsystem Weight

Weight of the pyrotechnic subsystem will not exceed 15.0 pounds.

8. Safety Considerations

The design of the pyrotechnic subsystem and interfacing subsystems will adhere to the requirements of the latest revision of AFETR M127-1, Safety, Range Safety Manual. The electromechanical pyrotechnic arming devices, viz, Separation Initiated Timer and Pyrotechnic Arming Switch, contain fail-safe arming status indication circuitry. The indication circuit is routed via the spacecraft umbilical connector to allow monitoring of the status in the launch complex blockhouse.

It is a requirement that no single or common failure mode will both arm and command a hazardous pyrotechnic event.

K. CABLING SUBSYSTEM

1. Description

The cabling subsystem hardware for the Viking Orbiter is changed from MM71 because the new orbiter configuration includes:

1) Four additional electronic assemblies
2) Larger bus structure and new adapters
3) Larger propulsion subsystem
4) Addition of capsule system with in-flight separation to capsule
5) Modification to scan platform support

The method of cabling and the type of connectors and wire used for MM71 will be used for Viking. Where possible, MM71 hardware will be used with minimum changes.

2. Adapter Harnesses

Adapter harnesses will have connectors at the field joints and in-flight separation connectors at the separation joints where the orbiter interfaces with the capsule adapter and the system adapter. The pin requirements for these in-flight separation connectors needs early definition to assist in determining their locations and possible need for connector development for this application.

3. Weight Estimate

The weight estimate for the cabling subsystem is listed in Table 8K-1.

Table 8K-1. Cabling Subsystem Weight Estimate

	Weight (lbs)
E. A. Harnesses	43.4(1)
Upper Ring	24.0
Lower Ring	7.0
Scan Platform	8.0
Spacecraft Miscellaneous Harnesses	14.5
Propulsion Module	12.0
	108.9

(1) Includes weight of support structure but balance of cable weights do not include weight of accessories for support to spacecraft structure.

L. PROPULSION SUBSYSTEM

The primary function of the propulsion subsystem for Viking is to provide directed impulses required for in-transit trajectory corrections, an orbit insertion maneuver at Mars, and two or more orbit trim maneuvers. The propulsion subsystem is comprised of a mechanically separable spacecraft module containing all necessary components and fluids to provide these impulses upon command. Maximum utilization of Mariner '71 technology and hardware has been the primary criterion in establishing the Viking propulsion design.

1. Propulsion Requirements and Design Constraints

Propulsion propellant requirements for the Viking mission are determined by the total impulse, which is dictated by the launch date, trajectory accuracy, desired Mars orbit and required orbit corrections, and by the orbiter and capsule weights, which are a function of desired engineering and science experiments and associated support system hardware requirements. The Viking propulsion system is sized for the following velocity requirements and spacecraft and capsule weights (firing schedule shown is approximate):

Midcourse correction ΔV	15 meters per second
First maneuver at launch + 15 days	
Second maneuver at encounter - 20 days	
Third maneuver at encounter - 18 days	
Orbit insertion ΔV (impulsive)	1350 meters per second
Gravity losses due to non-impulsive maneuver	110 meters per second
Orbit trim ΔV	
Orbit trims after encounter (prior to capsule separation)	50 meters per second
Second orbit trim at encounter + 10 days (after capsule separation) Additional trim maneuvers during orbiter 90 day mission	50 meters per second

Spacecraft Weight (excluding propulsion subsystem)	1382 pounds
Capsule Weight	1800 pounds

The effect of "off-loading" the propulsion system on spacecraft launch weight and ΔV capabilities is described in Appendix C of this document.

The propulsion system shall be capable of performing orbit trim maneuvers throughout the 90-day orbiter operational period; thus, a reliable propulsion subsystem functional lifetime of approximately 325 days is required.

Impulse-bit requirements, accuracy requirements, and engine thrust vector alignment and offset tolerances have not been determined to date. However, since the trajectory requirements are similar to Mariner '71 requirements, impulse-bit and accuracy specifications for the Mariner '71 propulsion system should be a reasonable guide for setting Viking specifications.

Envelope and arrangement of this subsystem must be compatible with interfacing structure and shall consider the potentially conflicting geometry requirements of other subsystems including antenna and folded solar panels, and view angle requirements for instruments and sensors. Heat transfer from the engine to the spacecraft through conduction, radiation or exhaust impingement shall be minimized, and location and design of the thrustor shall consider potential plume effects on adjacent surfaces, instruments, and on command and data transmission to and from the spacecraft. Direct impacts of propulsion design options on weight of the power system shall be considered in the propulsion tradeoff and selection process.

A primary consideration in the selection of hardware for the Viking propulsion subsystem shall be to utilize developed and qualified components wherever possible, thereby maximizing mission success, and minimizing program costs.

2. Subsystem Description

a. General

The propulsion subsystem is functionally a pressure-fed multi-restart, fixed thrust, storable bipropellant propulsion system, depicted schematically in Figure 8L-1, utilizing the propellants nitrogen tetroxide (N_20_4) and monomethylhydrazine (MMH), at a weight mixture ratio of 1.55 O/F. The propulsion subsystem includes all of the mechanical, structural, pneumatic, and hydraulic subassemblies required to provide a directed impulse from a mechanically-separable modular assembly. Principal subsystem components consist of a mounting structure, a high-pressure gas reservoir, a pneumatic pressure regulator, four propellant tanks with positive expulsion provisions, and a rocket engine assembly which is electromechanically gimballed in two planes and utilizes a direct acting, electrically operated, normally closed, linked, bipropellant valve. Electrical components and cabling required to transmit and respond to external excitation to direct or control impulse are included in the propulsion subsystem. Thermal control equipment (heaters, insulating blankets, etc.) and flight instrumentation are also included. The low-thrust, random demand, attitude control function that would be required during periods of celestially referenced cruise is excluded from the propulsion subsystem (see Section VIII-I of this document).

The design concept is predicated on the basis that the subsystem must satisfy the long-term space storage (maximum of 325 days) and multiple start (maximum of seven) requirements of the Viking mission. To better fulfill the long-term storage requirements, pressurization by gaseous nitrogen is utilized in lieu of gaseous helium, the latter being more prone to diffusion and leakage. Welded or brazed tubing and fittings are used whenever possible to minimize the number of potential leak sources. Metal seals will also be employed, where welding and brazing are impractical, in place of elastomeric seals to alleviate the effects of irradiation, hard vacuum, temperature, and long-term storage on the critical system seals.

Leakage will be controlled by the inclusion and selected use of multiple explosive valves arranged in two parallel branches of normally closed (NO) and normally open (NC) valve groups located in the pressurant

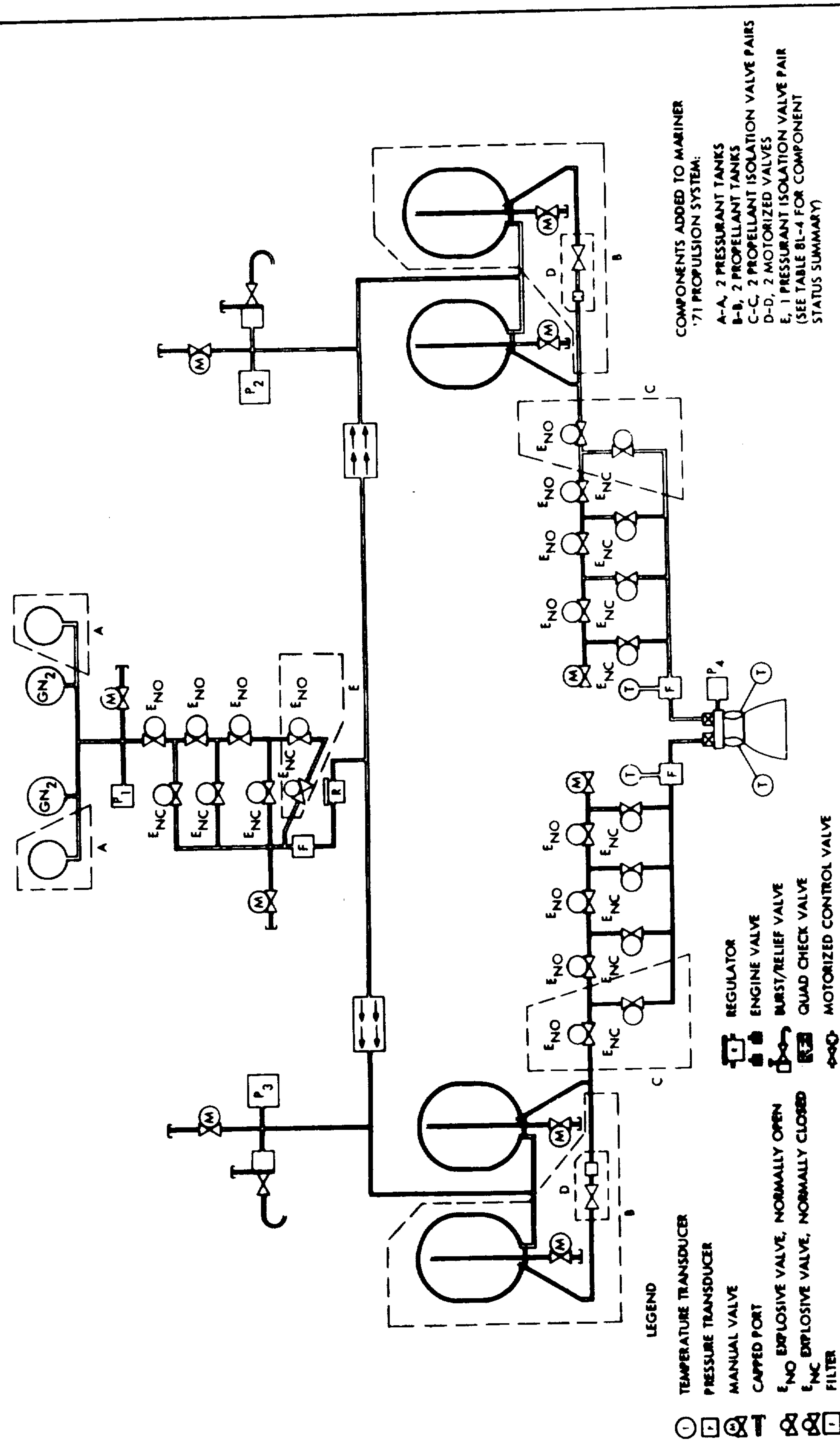

Figure 8L-1. Viking Propulsion Subsystem Schematic

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and propellant flow circuits. Prior to the first trajectory correction, the closed explosive valves provide positive sealing.

After the first correction, the intent is to rely upon electrically operated valves of the engine and intrinsic no-flow sealing of the gas regulator for short duration leakage control. When no other immediate corrections are anticipated, or if telemetry indicates a significant leakage, explosive valves will be used to provide positive leakage control.

The proposed propulsion subsystem is capable of being fueled, pressurized, and monitored for a minimum of 50 days prior to encapsulation of the spacecraft and actual launch. The subsystem in the pressurized and fueled condition will be safe for personnel operations and handling over a temperature range of +20° F to 105° F. No spacecraft umbilicals or hardlines are required to maintain the propulsion subsystem in the "ready" condition. Propellant fill and bleed fittings are located so as to provide for a single access from a shroud port should deservicing be necessary on pad.

Control commands to, and subsystem event sequencing for, the propulsion subsystem will originate from the spacecraft central computer and sequencer (CC&S). Storage of squib energy and management of solenoid power shall be accomplished by necessary power switching operations of the pyrotechnic subsystem. Continuous dynamic command inputs shall be provided to the propulsion subsystem during thrusting periods by the spacecraft autopilot. These commands will dynamically orient the thrust vector by means of the gimbal actuators.

b. Pneumatic Assembly

The pneumatic assembly includes four high pressure gas storage vessels with fill valve and flight instrumentation, four pairs of ganged explosive valves (one NC, one NO per pair), a single stage pressure regulator, a flow filter, flow check valves, a nd pressure relief provisions. Table 8L-1 provides further description of this assembly.

Table 8L-1. Propulsion Component Characteristics

Pressurization Assembly	
Gas	N_2
Tank material	Ti 6Al 4V
Tank diameter (in.)	18
Tank capacity (lb of N_2 in 4 tanks)	104
Pressures - psig	
Operating	4000
Proof	6600
Burst	8400
Gas flow control	single stage regulator with ganged pyrotechnic isolation valves, series parallel checks
Leakage rate	150 scc/hr. maximum (all paths)
Propellant Supply Assembly	
Propellants	N_2O_4/MMH
Tank material	Ti 6Al 4V
Propellant expulsion	Laminated teflon bladder with standpipe and screens
Tank dimensions (in.)	30 dia x 44 in. long
Tank capacity (lb of usable propellant in 4 tanks)	3130
Pressures - psig	
Operating	242
Proof	404
Burst	484
Propellant flow control	solenoid valve with ganged pyrotechnic isolation and backup valves
Relief pressure	330 psig
Engine Assembly	
Thrust - lbf	300
Specific impulse (ϵ = 40:1) - lbf-sec/lbm)	279 Minimum
Chamber pressure - psia	117
Mixture ratio (O/F)	1.55
Thrust vector control	Gimballed (2-axes)
Material	Beryllium chamber, L-605 nozzle

c. Propellant Supply Assembly

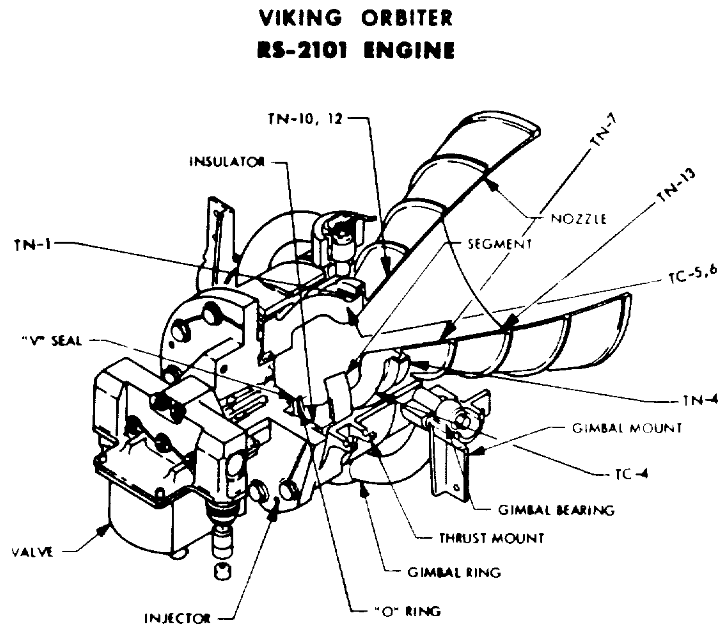
The propellant supply assembly includes four equal volume propellant tanks, each with internal teflon bladder and outflow standpipe and screens. Both pairs of intermanifolded tanks will be separated by motorized valves to eliminate propellant migration during coast periods. Also included are the ganged pyro valves in both the fuel and oxidizer circuits, as well as necessary fill, vent and bleed valve provisions. Propellant filters are included to control the influx of particulate contaminants to the engine propellant valves. Further description of the propellant supply assembly is included in Table 8L-1.

d. Engine Assembly

The engine assembly selected is a Rocketdyne RS-2101 engine currently being qualified for the Mariner '71 spacecraft. The engine utilizes a Rocketdyne proprietary "interegen" cooling concept for chamber cooling and a radiation cooled nozzle skirt providing a 40:1 expansion area ratio. This single 300-pound-thrust engine is required to thrust for approximately 3000 seconds during the orbit insertion maneuver. Interegen engines of 100 pounds thrust have been continuously fired for over 10,000 seconds, and since the RS-2101 reaches thermal equilibrium after about 200 seconds, no problems are anticipated in demonstrating the long duration burn capability. Thrust vector control includes two-axes pivoted gimballing with two push-pull linear electromechanical actuators. The bipropellant valve coupled to the engine is a dual-coil electrical torque-motor design. A cutaway view and dimensions of the RS-2101 engine are shown in Figure 8L-2. Temperature distribution through the RS-2101 chamber vs. time is shown by the data in Figure 8L-3; with locations as specified in Figure 8L-2. Nozzle exhaust cone temperatures, for steady-state operation, are indicated in Figure 8L-4.

e. Structure, Thermal Control and Cabling

The structural assembly, thermal control, and cabling of the propulsion subsystem are discussed in their respective sections of this document.



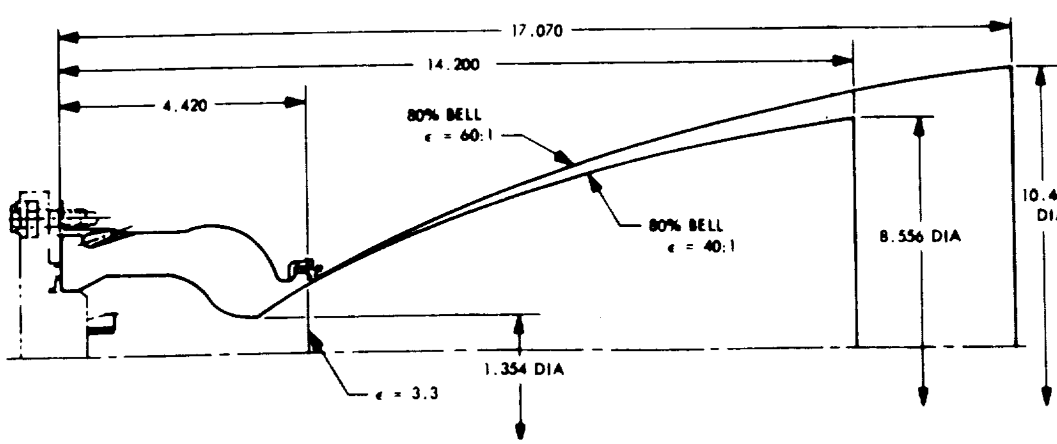


Figure 8L-2. RS-2101 Engine

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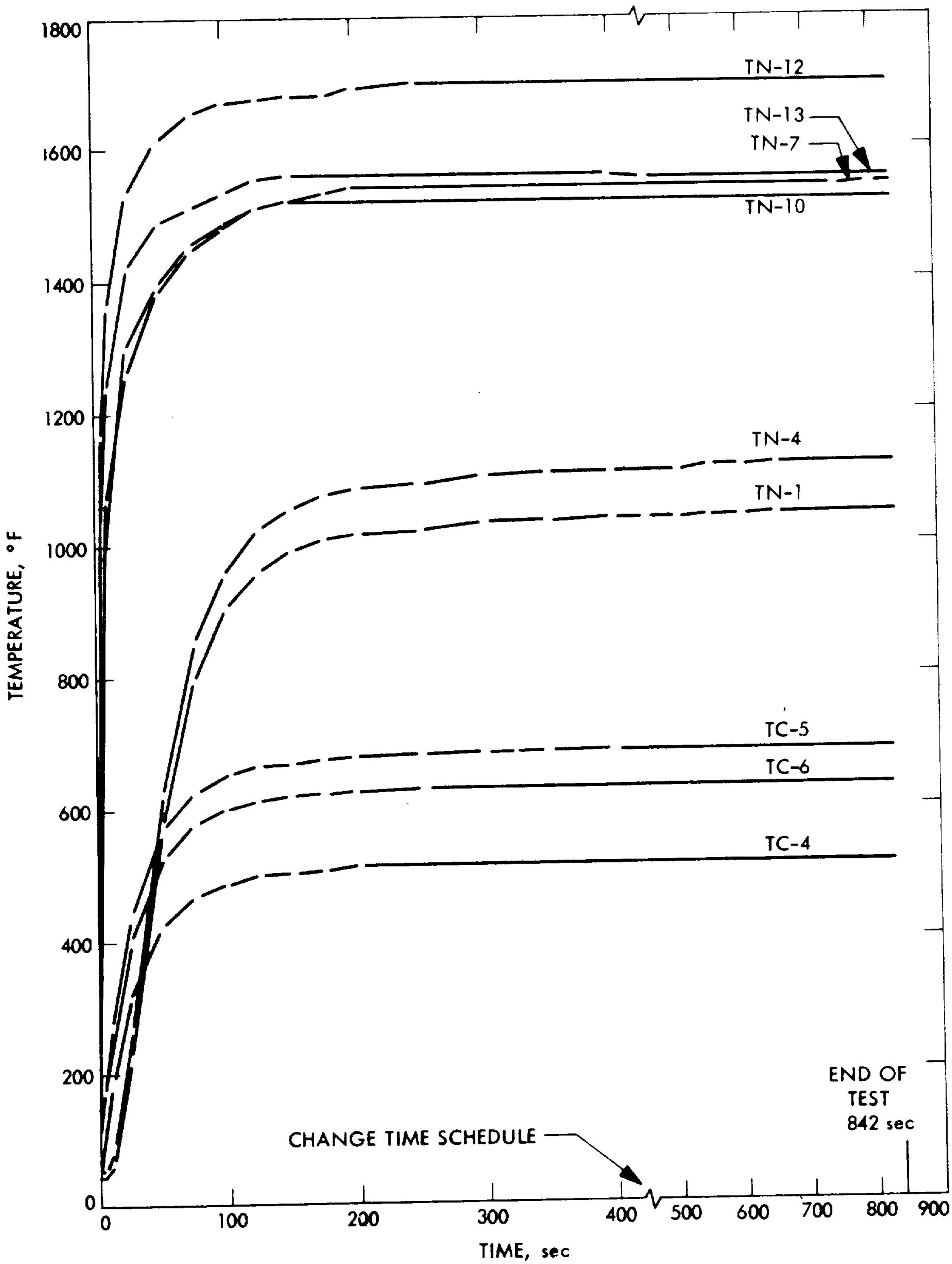

Figure 8L-3. Time-Temperature Characteristics of RS-2101 Chamber

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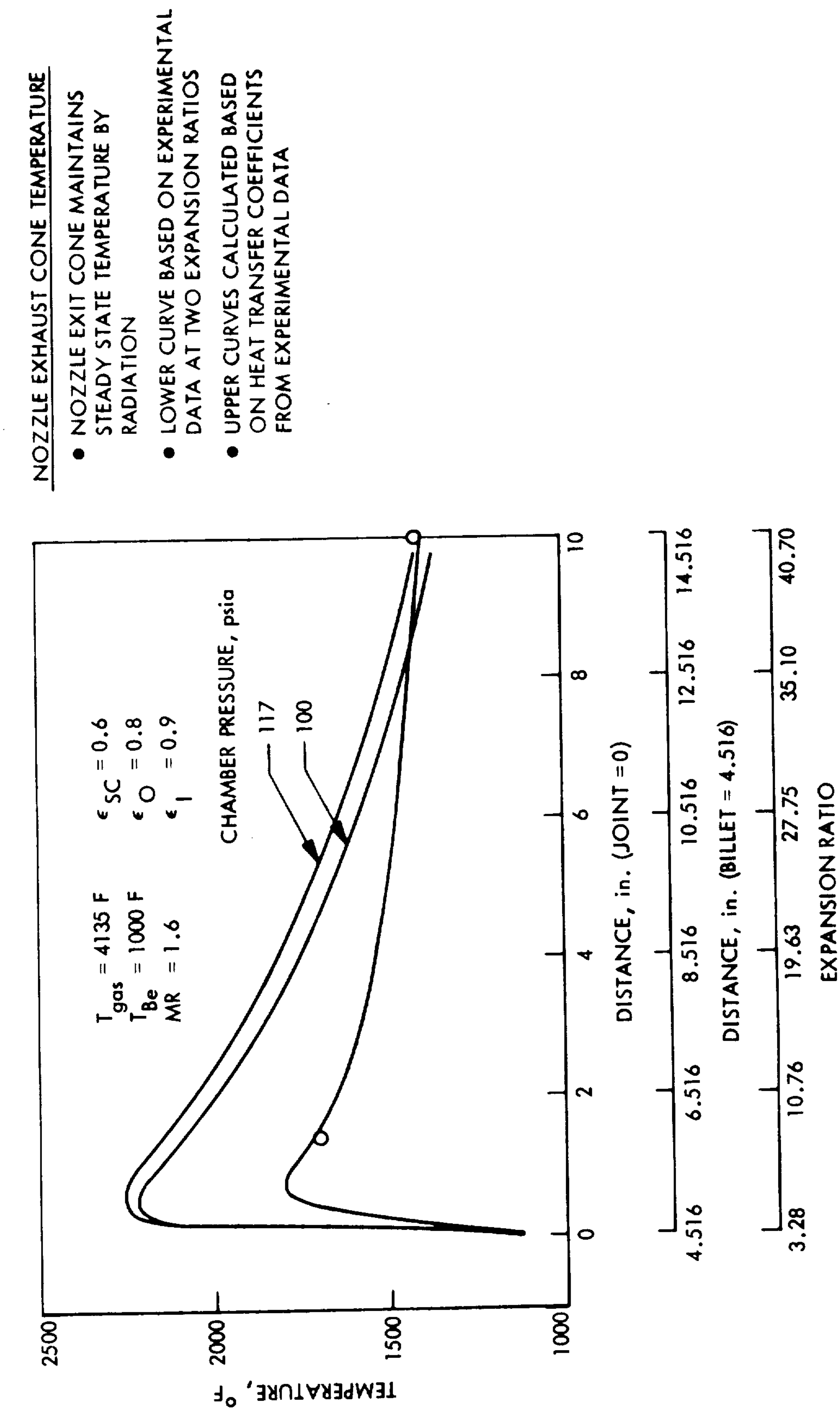

Figure 8L-4. RS-2101 Nozzle Exhaust Cone Temperature

f. Interface Characteristics

The interface characteristics of the propulsion subsystem are presented in Table 8L-2. An approximate exhaust plume map for the RS-2101 engine is presented in Figure 8L-5. It appears that, with the current Viking propulsion-subsystem/spacecraft configuration, there will be no dynamic-pressure or heating-rate problems associated with plume impingement on the spacecraft.

3. Performance, Impulse Accuracy, and Alignment

Nominal steady state specific impulse performance of the Viking propulsion system is 285 seconds. Instrumentation inaccuracies and manufacturing tolerances cause a possible variation from the nominal performance of 6 seconds, producing a minimum expected specific impulse of 279 seconds. An increase in nominal specific impulse of approximately 3 seconds can be gained by lengthening the nozzle to an area ratio of 60:1. The specified minimum specific impulse of 279 seconds has been used to establish propellant requirements for the Viking mission.

Minimum impulse-bit capability is dependent only on the engine shut-down reproducibility since the spacecraft is equipped with on-board accelerometers. A capability of better than 120±3 lb_f-sec can be achieved with the Viking propulsion subsystem.

Thrust level is 300±15 lb_f with a specified vector variation of ±0.5° in alignment and ±0.05 inches centerline offset between the geometric engine centerline and the actual engine thrust vector. Inaccuracies in mounting and aligning the engine within the propulsion module and spacecraft must be accounted for in addition to the above specified tolerance.

4. Weight

A weight summary for the baseline propulsion subsystem described above is presented in Table 8L-3. Total weight shown in this table includes an eight percent contingency on inert weights, and the propellant requirements are based on the velocity requirements specified in paragraph L-1 above and the

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Table 8L-2. Propulsion Interface Characteristics

Thrust Level	
Thrust	300 ±15 lb_f
Guidance and Control	
Thrust vector alignment to engine	±0.5°
Engine alignment	TBD
cg shift during propellant use	TBD
Roll torque	2 in.-lb (subject to review)
Gimbal cone of thrust vector	$\leq$ 13°
Thermal Control - Temperature Limits	
Pressurant system, prior to burn	+20 to +105° F
Propellant tanks and lines	+20 to +105° F
Engine valve-- pre-ignition/post shutdown	+20 to +105/<250° F
Injector - pre-ignition	TBD
Power Requirements	
Solenoid valve	20 w
Gimbal actuators	TBD
Instrumentation and Command	
Temperature channels	12 parameters
Pressure channels	4 parameters
Commands	TBD
Electrical monitors (gimbal position)	2 signal output

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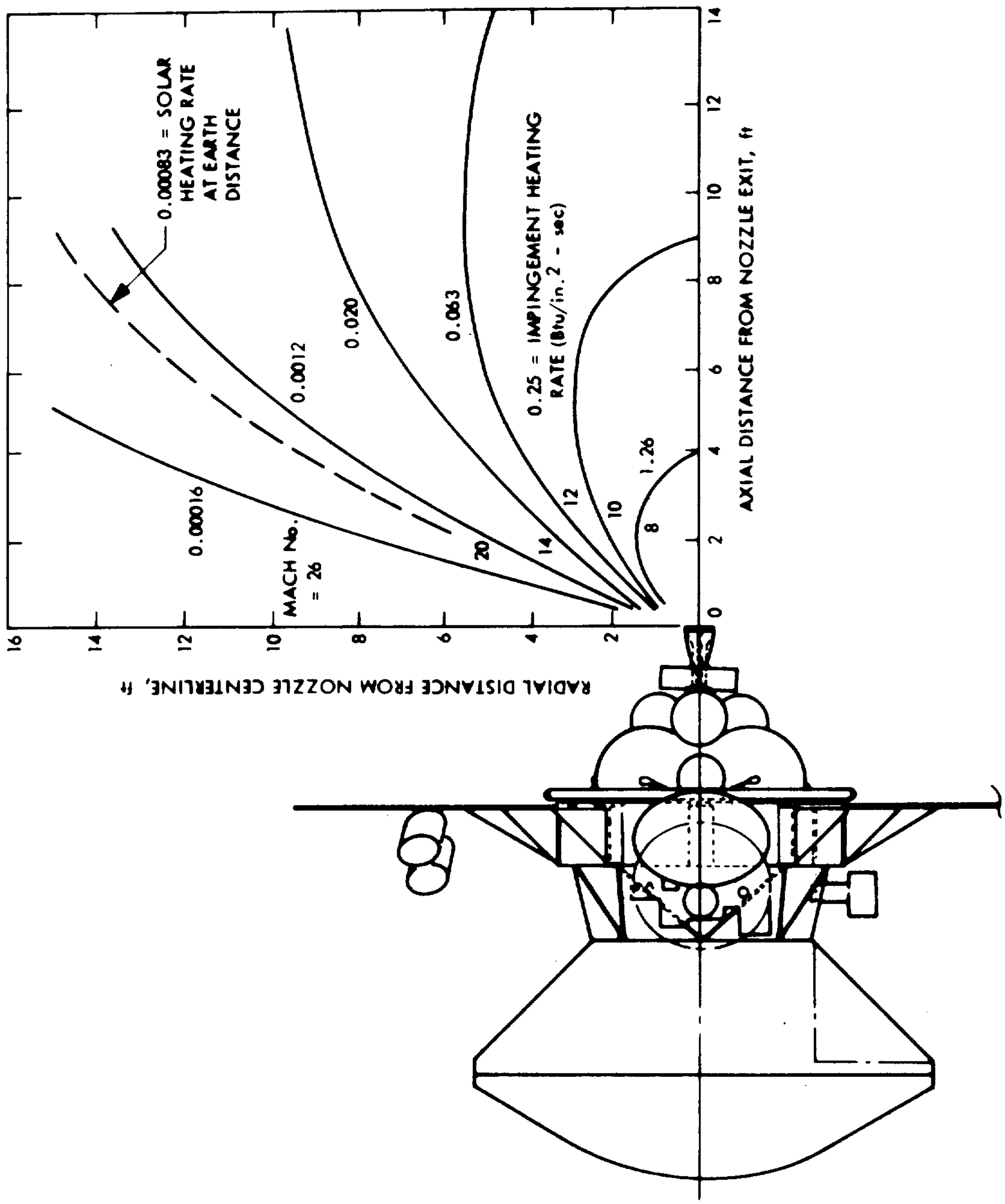

Figure 8L-5. Approximate Exhaust Plume Map for RS-2101 Engine

Table 8L-3. Viking Propulsion Subsystem Weight Summary

Rocket Engine Assembly Biprop valve, TCA, nozzle extension, etc.	18
Propellant Isolation Assembly Eight NO/NC pyro valve pairs and manifolding	32
Propellant Tank Assembly Tanks, bladders, standpipes and screens, inter-tank manifolding, and motorized isolation valves	203
Pressurant Check and Relief Assembly	14
Pressurant Control Assembly Four NO/NC pyro valve pairs and manifolding	20
Pressurant Tank Assembly	191
Nitrogen Pressurant	102
Propellant Residuals Provides for loading accuracy, entrapment, MR variations, and bladder expulsion efficiency	141
Command and Squib Harness	6
Structure	61
Subtotal	788
Contingency	62
Total Inerts and Residuals	850
Usable Propellants	3130
Total Propulsion Subsystem Weight	3980 pounds

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specified minimum specific impulse of 279 seconds

The weight estimates in Table 8L-3 were derived from actual components (described in paragraph L-5) or from scaling similar Mariner '71 components. The resulting propellant fraction, $\frac{W_{propellants}}{W_{propellants} + W_{inerts}}$ of 0.77* is typical for this propulsion system type and size.

The 3130-pound usable propellant-capacity of this baseline design allows for off-loading of propellants during non-optimum portions of the launch window (see Appendix C of this document).

5. Component Status

The Viking propulsion subsystem utilizes primarily existing components which are currently being qualified for the Mariner '71 program. Basic additions to the Mariner '71 propulsion system are noted in Figure 8L-1 and Table 8L-4. Due to the larger propellant volume requirement, the Viking propulsion system will use stretched versions of the Mariner '71 tanks, and extended bladders and standpipes using Mariner '71 technology.

6. Future Analysis and Decision Requirements

The most significant design decisions, in terms of overall system impact and timeliness, which must be resolved early in the Viking design activity are:

1) Engine selection -- the feasibility of the 3000-second burn capability must be demonstrated for the 300-pound thrust

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*If, as is commonly done, the structure related to the propulsion system, meteoroid shield, thermal control, and contingency weights associated with this hardware are not charged to the propulsion subsystem, the propellant fraction is increased to 0.81. Note also that 52 pounds of inert weight are contributed by isolation valving and 34 pounds by the standpipe/screens and bladders; both of which are dictated by extended zero-g coast periods, characteristic of the Viking mission.

Table 8L-4. Propulsion Subsystem Component Status Summary

Component	Source Program	Supplier
Propellant tanks	"Stretched" Mariner '71 tanks	(TBD)
Propellant bladder	Mariner '71 (extended)	(TBD)
Standpipe/Screen	Mariner '71 (extended)	Rocketdyne
Pressurant tank	Several suitable tanks are available	(TBD)
Fill valves	Mariner '71, MMBPS	JPL, Pyronetics
Explosive valves	Mariner '71	Pyronetics
Squibs	Mariner '71	(TBD)
Pressure regulator	Mariner '71, Minuteman (PBCS)	National Waterlift
Check valves	Mariner '71, Apollo	Accessory Products
Burst/relief assembly	Mariner '71, MMBPS, Apollo	Leonard, Calmec
Rocket engine Assembly	Mariner '71, PBPS	Rocketdyne

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beryllium engine. A demonstration full-duration firing of the RS-2101 engine is currently planned for the March/April '69 time period. Detailed verification of compatibility with power system, analysis of heat conduction and radiation during and after the extended firing time, and investigation of gravity losses and guidance techniques to reduce these losses, are also required.

2) Tank configuration -- selection and use of the four-tank arrangement will depend on detailed analyses of tank and structural weights, spacecraft envelope and packaging compatibility, propellant dynamics, c.g. drift and propellant management, and adequacy of bladder and screen/standpipe zero-g propellant acquisition devices.

3) Additional analyses related to the propulsion subsystem will be required in the following areas:

   a) General propellant acquisition and propellant management problems associated with the Viking mission including investigation of series and parallel options for the four-tank arrangement.

   b) Determination of pressurant tank location, number of pressurant tanks, tank structural mounting provisions, and relative reliabilities, leakage potential and hardware costs associated with the design options.

   c) Investigation of performance improvement techniques including increased nozzle area ratio, injector design improvements, and "optimization" of film-cooling requirements.

   d) Analysis of thermal control and meteoroid shielding requirements for the Viking propulsion subsystem.

   e) Tradeoff associated with propellant capacity and Viking launch-window improvement.

   f) Detailed failure mode and failure rate analysis of the propulsion system to establish mission critical failure effects and determine if the baseline system provides adequate redundancy and single-source failure protection.

g) Specification of Viking propulsion/spacecraft interface requirements.

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M. TEMPERATURE CONTROL SUBSYSTEM

1. Functional Description

The prime function of the thermal control subsystem is to maintain all equipment aboard the spacecraft within specified temperature limits. An active thermal control system is used for the internal orbiter equipment. Thermal control of equipment on the outside of the orbiter is achieved by the use of thermal coatings and thermostatically controlled electrical heaters.

2. Thermal Design

The thermal design of the orbiter is based on the concept of regulating the quantity of rejected heat from an insulated orbiter. The orbiter is insulated with multilayer superinsulation, using the louver system to provide regulated heat rejection. Since the electronic assemblies represents the bulk of the heat generated inside the orbiter, they are mounted directly to the baseplate radiator in front of the louvers. This results in minimum resistance thermal path. Maximum radiation and conduction coupling is provided inside the orbiter between all equipment bays so that portion of the heat in a high power bay can be transferred to the low power bays. This is also useful during the midcourse maneuvers where the Sun may incident directly into several of the equipment bays. Since the propulsion module, unlike the electronic assemblies, does not generate any heat during cruise, the system must rely on the equipment bays to transfer enough heat to keep the propellant tanks from getting too cold. In addition, two 10 watt heaters are provided to minimize any gradient around the tanks. A high temperature insulation blanket provides an enclosure for the propulsion module to provide protection during midcourse and orbit insertion firings.

Thermal control of equipment on the outside of the orbiter is achieved by passive means and/or by electric heaters. The solar panels, Sun sensors and other Sun oriented packages are controlled by the use of thermal control coatings in conjunction with controlled conduction paths to the orbiter structure. In addition, thermostatically controlled heaters are provided to compensate heat loss during Sun occultation.

3. Hardware Description

a. Thermal Louver

Thermal control of the main body of the orbiter utilizes a louver system composed of eight louver assemblies, each louver blade independently controlled by a bimetallic actuator which senses the local radiating panel temperature. As the panel temperature increases the louvers open, thereby increasing the effective emittance of the panel. Mechanical stops limit louver positions between 0° to 90° (fully open). The heat dissipation capability of the louver system is shown by Figure 8M-1.

b. Thermal Insulation System

The multilayer high performance insulation is used to cover the entire orbiter except for the required openings and penetrations (e.g. adapter, outrigger assemblies). The insulation if made of 25 layers of 1/4 mil aluminized Mylar with a 6 mil Armalon face sheet serving as part of the micrometeoroid protection subsystem for the orbiter. In the vicinity of the orbit insertion engine, high temperature insulation is used which consists of 24 layers 1/2 mil gold coated Kapton with a 6 mil Armalon and a 2 mil gold coated Kapton as the face sheet.

In addition, an 18 mil stainless steel collar is provided around the throat of the orbit insertion engine to provide protection to the gimballing assembly during engine firing.

4. Lander Capsule and Orbiter Interface

Conduction heat transfer between the lander capsule and orbiter is minimized by using low conductivity structural configuration for the lander capsule adapter. A truss with only four attachments to the orbiter was selected. Radiation heat transfer between the lander capsule and orbiter is minimized by configuration control, isolating the lander capsule from the orbiter.

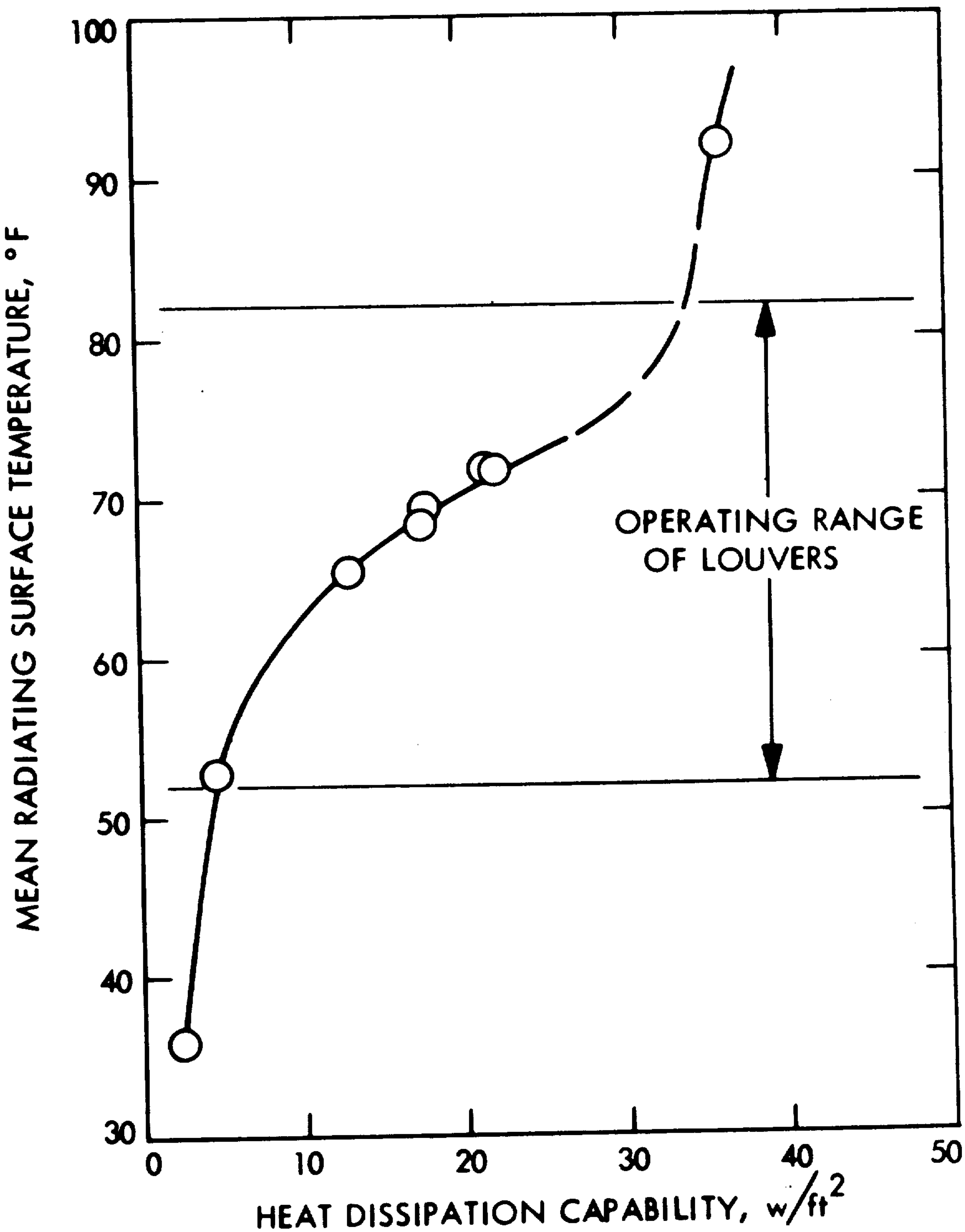

Figure 8M-3. Heat Dissipation Capability of Louver System

Launch pad air conditioning, provided by the launch vehicle, must be able to accommodate 600 watts power dissipation from the orbiter and any additional power dissipation by the lander. The temperature at the air conditioning inlet shall not be less than 40°F. The maximum heating rate from the shroud to the orbiter during the ascent phase is assumed to be 40 watts/ft^2. The shroud is ejected no earlier than 350,000 ft altitude so that free molecular heating of the orbiter shall not cause any major overheating problems.

5. Weight Estimates

The weight estimates for the temperature control subsystem are listed in Table 8M-1.

Table 8M-1. Temperature Control Subsystem Weight Estimate

	Weight (lbs)
Scan Platform	2.50
Insulation - 4 bays	1.00
Upper Insulation Blanket	2.00
Maneuver Motor Heat Shield	1.20
Propulsion Module Insulation	5.40
8 Louver Assemblies	10.50
Bay Corner Shields	1.50
Louvers for Motor Thrust Plate	1.00
Channels (Louver Bay @ .1/bay)	0.80
Miscellaneous (heater, etc.)	1.10
	27.00

N. MECHANICAL DEVICES

1. Solar Panel Devices

The solar panels are hinged on outriggers attached to the bus. They are folded down during launch and are restrained by the spacecraft adapter. The panels must separate from the spacecraft adapter to effect the spacecraft separation from the launch vehicle. This is accomplished with a device which incorporates a bayonet-socket type tiedown with dampers in the support linkage to attenuate panel loads during vibration.

The panel erection actuator springs and the cruise damper-locking mechanization are similar to those used for Mariner '71 which has similar panels supported on outriggers.

2. Antenna Deployment Mechanisms

a. High Gain Antenna

The 58" diameter antenna reflector must be stowed during launch to fit within the available dynamic envelope. After injection the antenna is deployed to its initial position by an open loop deployment assembly. Daily antenna update is required to give the maximum performance for the antenna. However, the pointing angles are such that this can be accomplished by rotation about only one axis.

b. Relay Antenna

The relay antenna array is mounted on a solar panel beam structure to give the required antenna pattern. The size of the array prohibits a fixed mounting position -- constrained by the available dynamic envelope. Therefore, the array is stowed during launch, latched by a pin puller, and is deployed by rotating it to its required attitude by a spring actuator. This technique was used on the MV67 and MM71 high gain antennas.

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3. Low Gain Antenna Support

The low gain antenna which covers the hemisphere on the Sun side of the spacecraft is typical to that used on MM64 through MM71. The antenna is supported on a long tube which serves as a wave guide to the antenna. This is base mounted to the bus structure. The tube is supported, during launch, near its 3/4 point by a damper support assembly attached to the propulsion module. This stowed position gives the required clearance for the separation maneuver. After separation, the antenna is deployed to a position parallel to the roll axis and is supported by a cruise damper support mechanism. This location positions the antenna as far from the radiatively cooled maneuver motor as is practical for this configuration.

4. Separation Mechanization

The spacecraft is attached to the spacecraft adapter at four points. The attachment is made by a pyrotechnically initiated device such as an explosive bolt. Separation velocity is achieved by the use of four springs housed on the adapter and pushing against pads on the lower ring of the orbiter.

A similar, but smaller assembly, is used to jettison the lander capsule adapter and bioshield base after capsule release.

5. Scan Platform

The scan platform definition is not yet complete as requirements are not fully known but its design is similar to Mariner '71. It is located on the roll axis and between the orbiter and capsule -- a location similar to MM71. Because the propulsion module occupies the center volume of the bus, the scan platform bearing supports are moved. This permits a good spacing for the bearings and also increases the pointing cone angle, somewhat, into the range less then 90°.

A 90° cone angle is required to accomplish the landing site surveillance required prior to capsule release.

Stowage will require a latch mechanization different from MM71 because the larger bus and location is not compatible with direct latching to the bus structure. The latching assembly will be designed to resist motions about the cone and clock actuation axes, thus keeping launch loads off the actuators. They must, however, be capable of taking orbit trim loads.

6. Devices Weight Estimate

Table 8N-1 lists weight estimates for the devices subsystem.

Table 8N-1. Devices Subsystem Weight Estimates

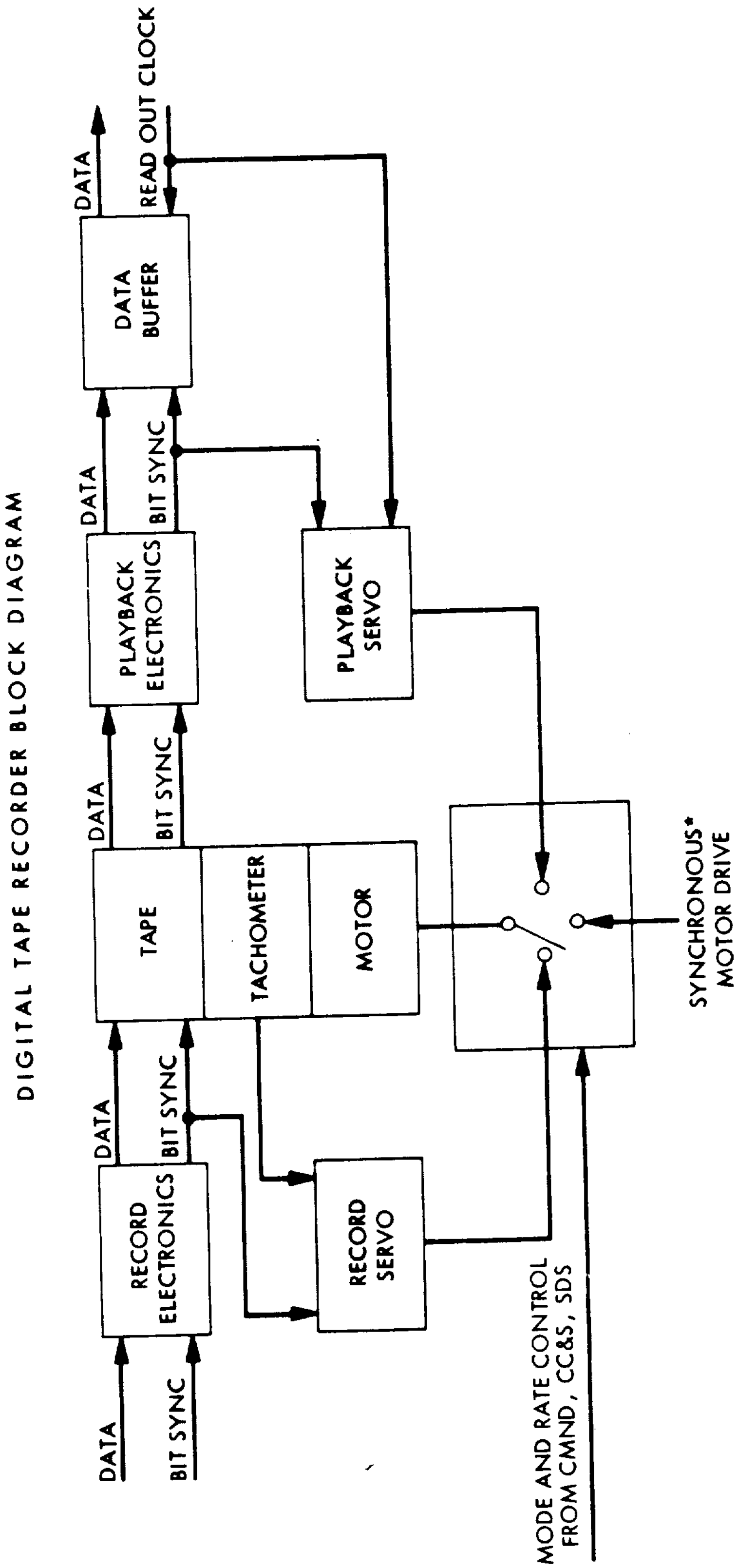
		Weight (lbs)
Solar Panel Devices		8.4
Launch Latches and Dampers	5.0	
Fold-out Locks and Cruise Dampers	1.9	
Deployment Assembly	1.5	
Antenna Devices		3.4
Low Gain Damper Assembly	1.4	
High Gain Latch & Deployment Mechanism	1.5	
Relay Antenna Deployment Mechanism	0.5	
Scan Platform		60.0
Frame Assembly, Bearings	45.0	
Latching Assembly	15.0	
Separation Initiated Timer		0.7
Pyro Arming Switch Assembly		0.1
Total		72.6

O. DATA STORAGE SUBSYSTEM

The orbiter data storage subsystem consists of two identical digital tape recorders and their associated record, playback, and control electronics. Figure 8O-1 is a block diagram of one of the recorders. The storage capacity of each recorder is 1.8×10^8 bits on four data tracks. There is a clock (bit sync) track associated with each data track. Playback rates are at 16.2, 8.1, 4.05, 2.025, and 1.0125 Kbps. Playback speed is controlled by means of a servo which uses the recorded clock and a read out clock, and the recovered data is shifted through an output buffer to remove jitter. Record rates are 132.3 Kbps for recording orbiter science data and 20 and 1.348 Kbps for recording capsule relay data. The tape recorder motor will be driven synchronously when recording 132.3 Kbps and in the servo mode at the lower bit rates.

DSS Parameters:

Weight	= 20 lbs each recorder
Power	= 22 watts each recorder
Storage Capacity	= 1.8×10^8 bits serial NRZ
Record Rates	= 132.3 Kbps, 20 Kbps, 1.348 Kbps
Playback Rates	= 16.2 Kbps, 8.1 Kbps, 4.05 Kbps, 2.025 Kbps, 1.0125 Kbps



*SYNCHRONOUS DRIVE USED FOR HIGH RATE (132 kbps) ORBITER SCIENCE RECORDING

Figure 8O-1. Digital Tape Recorder Block Diagram

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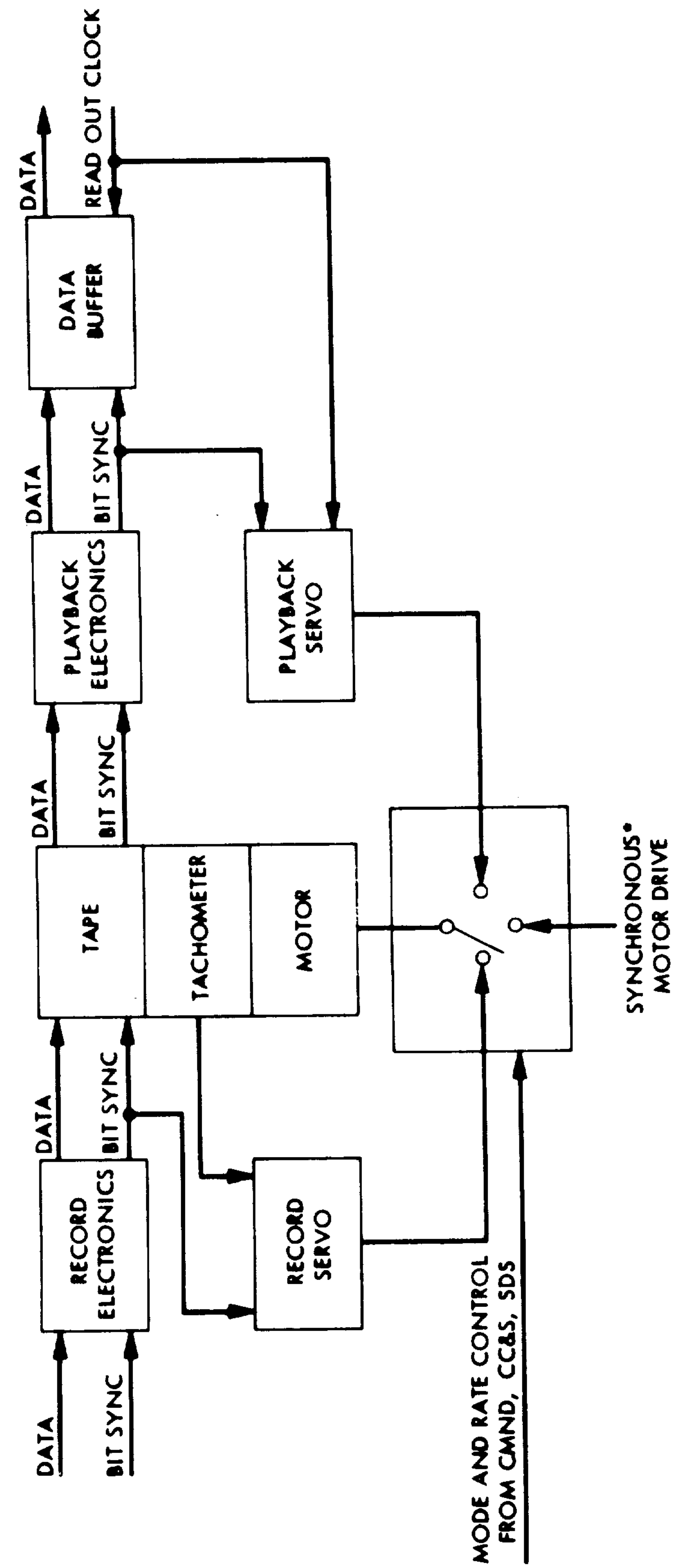

*SYNCHRONOUS DRIVE USED FOR HIGH RATE (132 kbps) ORBITER SCIENCE RECORDING

Figure 8O-1. Digital Tape Recorder Block Diagram

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P. SCIENCE DATA SUBSYSTEM (SDS)

The Science Data subsystem (SDS) serves as the data handling system for the science payload. It controls and sequences the science instruments, samples and converts the resulting data into digital form, provides temporary (buffer) storage and formats and routes the data to both the flight telemetry subsystem (FTS) and data storage subsystem (DSS) for direct or delayed transmission to Earth. The SDS exchanges commands with other subsystems aboard the spacecraft which pertain to the operation of the science payload.

The major differences between the Viking Orbiter and the Mariner '71 DAS are due to the increased flexibility required in sequencing the instruments and in constructing formats. This flexibility is required in order to sequence instruments in manner which will make the best use of information obtained from previous spacecraft orbits and the Lander.

The SDS will produce three data streams. A high rate and a low rate stream will be provided to the flight telemetry subsystem for transmission during the periods of recording and playback. The high rate will be used when a 210-foot antenna is available and the low rate at other times. The high rate will be progressively lowered from 16.2 Kbps to 8.1 Kbps to 4 Kbps as transmission conditions get worse during the progress of the mission. The low rate will be about 133 1/3 bps.

A 132.3 Kbps data stream will be sent to the digital recorder. This will consist of all TV and science data. The 16.2 Kbps and 8.1 Kbps data streams will consist of all science data and selected TV picture elements. The 4 Kbps and 133 1/3 bps will have selected portions of science and TV. Flexible formats will be particularly valuable in making optimum use of these lower data rates. Figure 8P-1 shows a gross functional block diagram of the SDS.

As the total science data rate is lower than in the Mariner '71 design, a smaller portion of the recorded data stream can be assigned to this data. This means that the TV can use a shorter flyback time if this is desirable. Total science data and coding will be no more than 350 bits. 450 bits were required in the Mariner '71 design.

The science data must be buffered during the time that TV video data

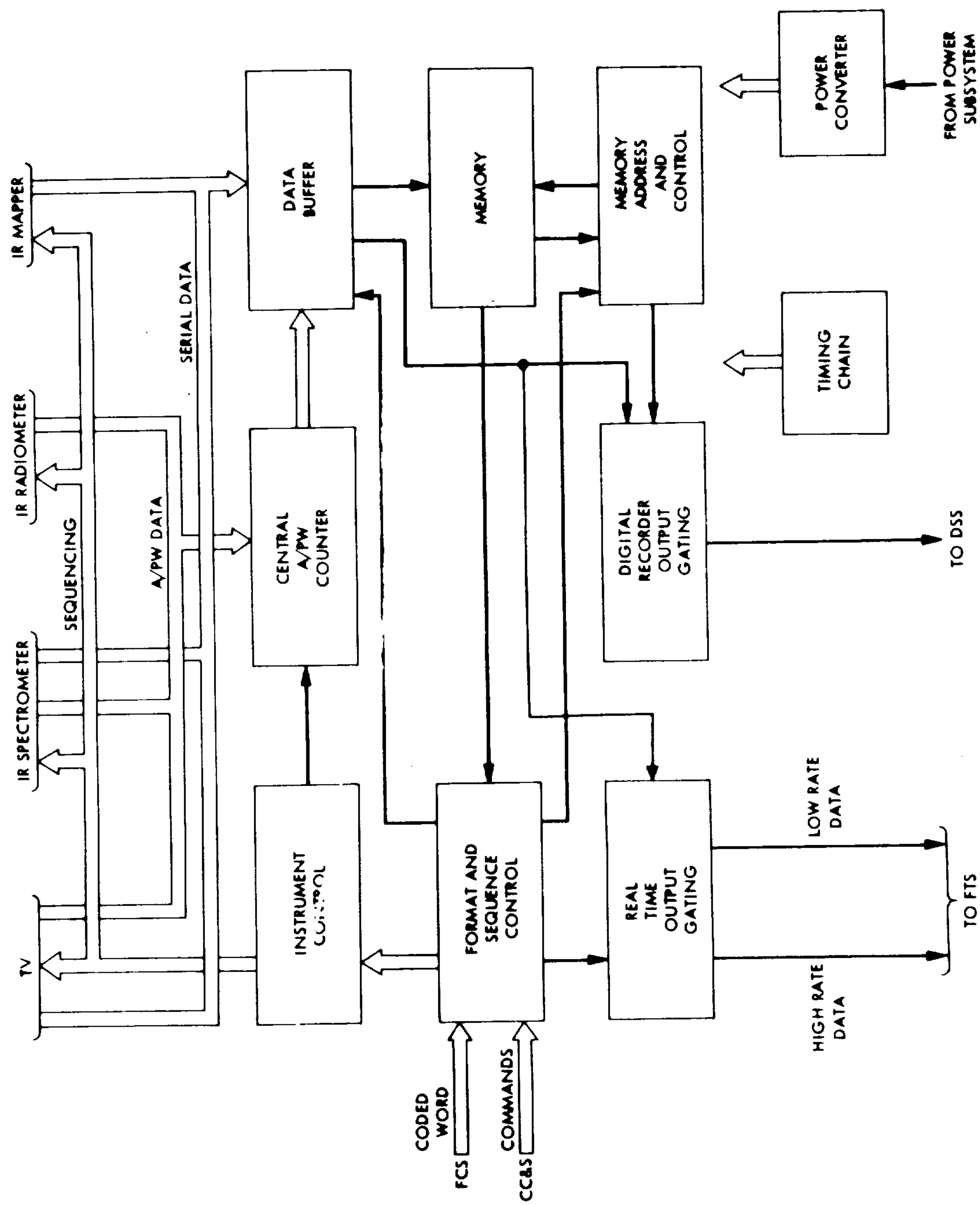

Figure 8P-1. SDS Functional Block Diagram

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is being recorded. A core memory will be used for this as in Mariner '71.

The core memory will also be used for storage of bit patterns used in controlling formats and sequencing.

A third use of the memory will be in the implementation of a non-volatile frame count. This will guarantee continuity of science data throughout the 60 or more days of the orbital mission.

The volume required for the SDS will be approximately 600 cubic inches. The power will be about 23 watts and the weight about 21 pounds.

Q. SCAN CONTROL SUBSYSTEM

1. Introduction

The Viking orbiter scan platform provides pointing for those scientific instruments which require orientation. The platform is a two degree-of-freedom structure located in the +z direction hemisphere of the spacecraft between the orbiter and the capsule. The clock axis allows rotation of the platform about the z axis of the spacecraft. The cone axis allows rotation of the platform about a line in the x-y plane. The cone angle is the angle between the platform line of sight and the spacecraft -z axis. The two degree-of-freedom motion of the scan platform is servo controlled by the scan control subsystem.

2. Functional Description

The Viking orbiter scan control subsystem consists of the electromechanical and electronic devices necessary to provide pointing control of the scan platform during the orbital phase of the mission.

The orbital phase of the mission may be divided into the pre-separation mode and the post-separation mode. The operation of the scan platform during these modes is described below:

a. Pre-Separation Mode

The available look angles of the scan platform prior to separation of the capsule lander are limited in cone by the presence of the lander in the +z direction hemisphere of the spacecraft. Therefore, in order to conduct landing site surveillance, the spacecraft will perform commanded turns to reposition the x-y plane of the spacecraft parallel to the orbital plane.

Rotation of the scan platform about the clock axis will then allow adequate TV coverage of proposed landing sites. The scan platform will be rotated about the clock axis by CC&S. The scan platform can also be controlled by radio command.

A block diagram of a single axis of the scan platform control subsystem is shown in Figure 8Q-1.

b. Post-Separation Mode

The function of the scan platform control subsystem after capsule separation is to position the directional science payload during orbital operations. The scan platform will move in both clock and cone in order to accomplish this task. The position of the platform will be controlled by a stored program in the CC&S. Both the clock and cone position of the scan platform can also be controlled by radio command. The minimum incremental movement of each axis is one degree for both FCS and CC&S commands.

3. Mechanization Description

The scan platform mechanization is detailed in block diagram form in Figure 8Q-1. The principal components in the scan control subsystem include the following:

1) Stepper motor drive electronics
2) Stepper motor and reference potentiometer assembly
3) Gimbal actuator pre-amplifier and power amplifier
4) Gimbal actuator and position potentiometer assembly

a. Stepper Motor Drive Electronics

This circuitry accepts pulses from either CC&S or FCS in order to actuate the stepper motor. Clockwise or counterclockwise rotation of the stepper motor may be commanded by either CC&S or FCS. The nominal pulse length of input commands is 100 milliseconds. To ensure noise immunity, the stepper motor drive electronics rejects input signals of less than 15 milliseconds duration. The electronics accepts an input signal and converts it to four sequential pulses at the stepper motor. Each output pulse drives the stepper motor 90 degrees or a total of 360 degrees for each group of four output pulses.

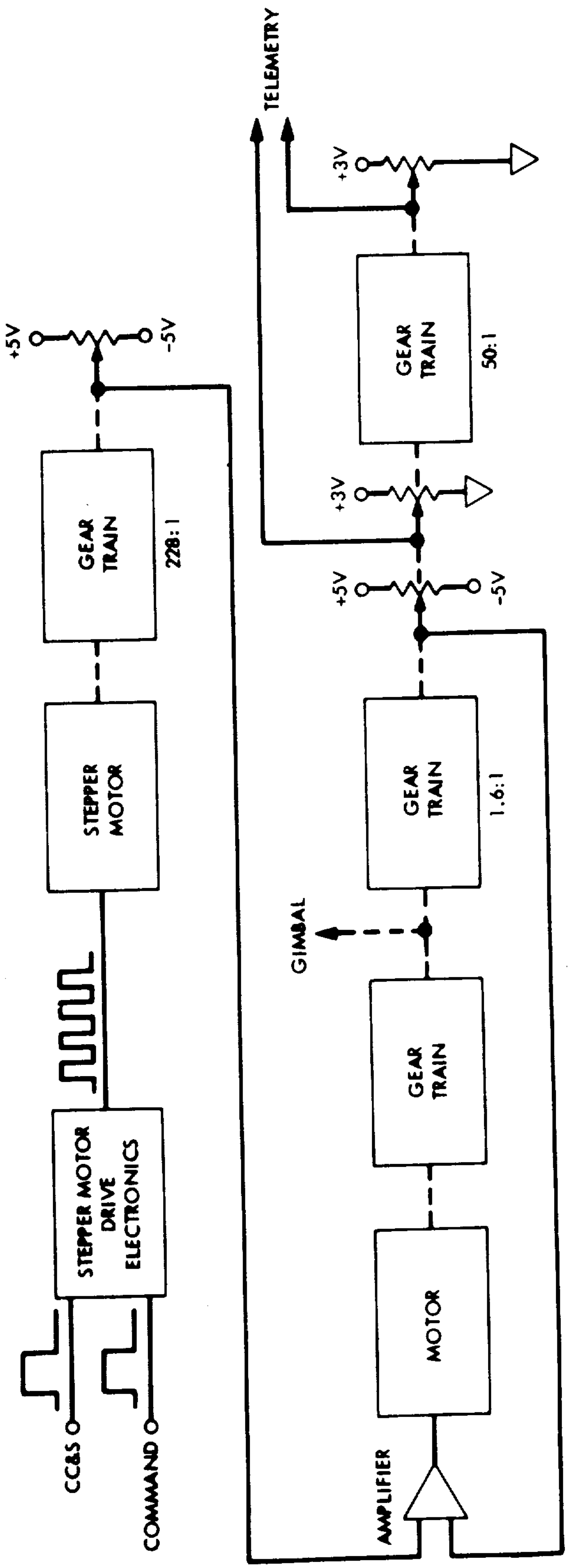

Figure 8Q-1. Single Axis Block Diagram for Scan Platform Control Subsystem

b. Stepper Motor and Reference Potentiometer Assembly

This assembly contains a 90°-per-increment stepper motor, a 228-to-1 step down gear train, and the reference potentiometer. For every 360-degree revolution of the stepper motor, the reference potentiometer moves approximately 1.6 degrees. The reference potentiometer has a useful range of 345 degrees. When converted to gimbal position, this represents approximately 215 degrees of possible gimbal movement per axis. However, spacecraft geometry will limit this travel to approximately 90 degrees in cone.

c. Gimbal Actuator Pre-Amplifier and Power Amplifier

The pre-amplifier amplifies the DC voltage difference between the reference potentiometer and the gimbal position potentiometer. As long as a difference exists, the pre-amplifier will have an output and will drive the gimbal power amplifier. The power amplifier is a 400-cycle, single phase amplifier that drives the control winding of the gimbal actuator motor.

d. Gimbal Actuator and Position Potentiometer Assembly

This assembly houses the gimbal actuator motor, 3 gear trains, the position potentiometer, and the coarse and fine telemetry potentiometers. A 400-cycle, 2-phase actuator motor is used to drive the scan platform gimbal. The motor uses a 26-volt 400-cycle fixed phase and a variable amplitude 400-cycle control phase. The control phase voltage is generated by the power amplifier and either leads or lags the fixed phase by 90 degrees depending on the desired direction of motor rotation. The output shaft of the gimbal actuator is connected to the gimbal position potentiometer by a 1.6:1 step-up gear train. A coarse telemetry potentiometer is located on the same shaft with the position potentiometer. A fine telemetry potentiometer is connected to this shaft by a 50:1 step-up gear train.

4. Pointing Accuracy Analysis

The scan control subsystem must point the scan platform instruments at the planet at various times during the mission. After the lander has been separated from the orbiter, this can be performed while the orbiter is Sun and Canopus oriented. In this case the expected 3σ values of the TV camera pointing error are 1.16 degrees in cone and 1.19 degrees in cross cone.

Prior to lander separation the pre-separation surveillance phase of the mission will be performed with the spacecraft under inertial control. A roll-pitch-roll commanded turn sequence will be performed to allow the scan platform instruments to see the spacecraft track on the planet surface. The expected 3σ values of the TV camera pointing error in this case are 1.53 degrees in cone and 1.55 degrees in cross cone initially. Due to gyro drifts, however, the scan platform pointing error will increase with time, in this case.

5. Weight and Power Summary

a. Weight

Listed below are the weights of the major components in the scan platform system.

	Weight (lbs)
Gimbal actuator ass'y	7.82
Mode and drive logic	1.44
Scan power supply	2.56
Servo amplifier and stepper motor ass'y	4.88
Total	16.70

b. Power Consumption

2.4 Hz: Peak, 28.5 watts
Nominal, 8.0 watts

400 cycle 1ϕ: Peak, 12.0 watts
Nominal, 10.0 watts

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6. Additional Considerations

Modifications to the existing hardware (MM71) might be considered in order to accomplish the following:

1) Decrease the minimum incremental resolution of the scan platform angular movement. It is now one degree in each axis.
2) Modify the electronics in order to operate an additional reference potentiometer. This potentiometer (not shown in the block diagram) can now (MM71) only be addressed by a prelaunch command while the spacecraft is on the launch pad. There is one additional reference potentiometer per axis in the present MM71 system.

There is a possibility that the scan platform may be located off the spacecraft z axis. This is not desirable due to adverse effects on the autopilot and effect on CG location. Since the engine thrust vector is nominally along the z axis, a substantial lever arm may exist when the platform is unlatched. The result is that severe dynamic interaction between the scan structure and autopilot control system may occur. If this is the case a reexamination of this problem will be required.

R. SCIENCE

In order to respond to the scientific objectives outlined in Section IV and to proceed with the conceptual design of the Viking orbiter, a model payload for the Viking orbiter mission was selected. This orbiter payload consisted of four science instruments, television, infrared radiometer, infrared interferometer spectrometer, and an infrared mapper. These instruments are discussed in some detail in the subsequent paragraphs. A summary of the model payload for the Viking orbiter is presented in Table 8R-1.

1. Television Subsystem

a. Basic Parameters

The television subsystem provides a photometric mapping of a large portion of the Mars surface, including the region from 10°S to 30°N latitude. In doing so, it forms a primary part of the scientific payload. The wide angle TVS field of view is a rectangle 11 deg x 14 deg, the longer dimension normal to the flight path. Assuming a periapsis speed for the orbiter of 4 km/sec and periapsis altitude of 1000 km, the field has an angular motion of .004 radians/sec or approximately 10°/frame of 42 sec. Further, with an orbit of 40° inclination, stepping the orbit between successive periapsis passages by 20° longitude (by changing the period to the period of Mars' rotation plus or minus 80 minutes) provides slight overlap for adjacent wide angle picture swaths. In order to generate narrow angle overlapping picture sequences at periapsis pass, the scan platform must be rotated between pictures. For a field motion of 4 microradians/millisec (given above) at periapsis, and a TV line separation of 25 microradians, an exposure time of 6 milliseconds or less is required to prevent line smear of less than one TV line.

b. Major Functional Elements

1). Camera A Optics -- The Camera A optics (wide angle) subassembly consists of a several-element

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Table 8R-1. Model Viking Orbiter Payload

	Weight (lbs) Total	Wt. (lbs) on Scan Plat.	Power Watts	Data (Kbps) Operating (Peak)
Television	48	36	38	132
High Resolution Infrared Radiometer	15	10	5	1
Infrared Interferometer Spectrometer	35	23	14	2.7
Infrared Mapper	30	15	14	0.3
Science Data Subsystem (SDS)	21	0	23	
Payload Total	149	84	94	136

refractor lens, and a shutter-filter mechanism. The shutter is a rotating disk-type with an appropriate filter introduced into the light path by the application of a current pulse. The effective focal length is 50 mm, and the focal ratio is f/3.0. The image tube raster affords a field of 11° x 14°.

2). Camera B Optics -- The Camera B (narrow angle) optics subassembly consists of a catadioptric telescope, and a shutter mechanism. The shutter is a solenoid-operated, dual-moving blade type. The focal length is 508 mm and the focal ratio is f/2.5. The field is 1.1° x 1.4°.

3). Camera Heads -- The camera heads are identical for both cameras and house the vidicon sensor, yoke assembly, preamplifier and distribution circuitry. These are mounted on the two-degree of freedom scan platform.

4). Signal Chain -- The signal chain processes the video signal from the camera heads and converts it to 9-bit digital information. The total video information per frame is 5.25×10^6 picture bits which must be stored on tape in the data storage subsystem. This requires a readout rate from the television subsystem in the neighborhood of 132 kilobits/sec.

5). TV Driver -- The TV driver generates the required sweeps for the camera heads. The sensor resolution affords 832 active picture elements/line, and 700 active scan lines/frame. Total frame time is 42 seconds.

6). TV Control -- The TV control distributes the required logic control to the various parts of the subsystem and provides the real time A/PW output.

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c. Special Requirements

1). The science data subsystem will sample one-half of each television picture frame and use this information to set the shutter speed for the next picture taken with that specific camera. In addition, the SDS will be capable of setting the shutter speed on ground command.

2). The SDS, on ground command, pre-condition the camera shutters for TV power turn off.

d. Interfaces

1). Mechanical -- The Camera A is 9" x 5" x 4" and contributes 8.75 lb on the scan platform.

Camera B is 32" x 10" x 10" and contributes 27.0 lb on the scan platform.

The spacecraft bus electronics is estimated at 16.7 lb for five blivets.

2). Electrical -- Primary power of 30 watts average is required not including temperature control power for temperature control of the Camera B optics. Heaters in the TV cameras will dissipate 3.5 watts of unregulated DC power. Heaters in the Camera B optics will dissipate 5.5 watts of unregulated DC power.

3). Thermal -- The portion of the television subsystem in the bus shall operate over a range of -25° C to +75° C. The camera assemblies shall operate from -20° C to +30° C. This subsystem shall survive a non-operating range of -30° C to +55° C.

2. High Resolution Infrared Radiometer (HRIR)

The HRIR provides a thermal map of the Mars surface, and in conjunction with the TVS, determines the features of any clouds that might be observed.

It contains a radiation cooled photoconductive cell and operates within the 1 to 5 micron region, and nominally has a noise equivalent temperature difference of 1°C for a 250°K background. Detector cooling is accomplished by means of a cooling patch at the bottom of a highly reflective gold-coated pyramid which is oriented to view cold space during the entire orbit.

The HRIR has an instantaneous field of view of .5°. A scan mirror continuously rotates the field through 360° in a plane normal to the spacecraft velocity vector. Radiation reflected from the scan mirror is chopped at the focus of a 4 inch f/1 modified cassegrain telescope. It is then refocussed at the detector by a reflective relay which contains a spectral filter. At a mirror rotation rate of 30 rev/min, continuous scanning occurs at periapsis The detector views a portion of space and the radiometer housing during every mirror rotation, the space scan serving as the zero reference, and the housing scan as a known temperature reference.

The radiometer video output is digitized by A/PW converter in the instrument and fed to the SDS central A/PW counter.

3. Infrared Interferometer Spectrometer (IRIS)

The IRIS has as primary objectives the obtaining of information on the vertical structure, composition and dynamics of the atmosphere and the emissive properties of the surface of Mars. The spectrometer will view the Mars atmosphere at discrete positions along the orbital path.

At the operating temperature of 250°K, the instrument generates an interferogram such that the IR power spectrum from 200 cm^{-1} to 1600 cm^{-1} (50 to 6 microns) can be resolved to at least 1.2 cm^{-1}. The Noise Equivalent Radiance is nominally less than 3×10^{-8} watts cm^{-1} $ster^{-1}$ throughout the specified spectral range at the operating temperature.

The field of view has a half angle of approximately 2.2°, at which radius off-axis the response is one half the on-axis response. A scanning interferometer mirror moves through at least .426 cm during the output IR sampling period of 18.2 seconds, done consistently with a 225 word/sec stream of 12 bit words, or a total of 4096 words per interferogram. The mirror

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drive system is required to keep the mirror velocity constant within 1% during the sampling period.

Image motion compensation is provided as a small rotation of an IMCC mirror to enable the same gound spot to be sampled during the entire interferogram.

At the completion of seven interferograms the IMCC mirror rotates 90° to allow an interferogram to be taken of interstellar background. At the completion of 15 interferograms it rotates 90° to view an onboard black body for one interferogram.

The science data is clocked into the SDS in serial form.

4. Infrared Mapper

The IR mapper has, as a primary purpose, the determination of presence and variation of atmospheric water vapor. The IRM will also determine the vertical variation of temperature in the atmosphere, the surface altitude variations, pressure variations and seasonal variations in the amount of CO_2, and pressure and temperature of clouds.

The optical design is basically an Ebert system with a fixed grating. The field of view at 1000 km is 5 km x 5 km for the PbS detectors and 75 km x 15 km for the PbSe detectors. The former detectors are used between .8 and 2.85 microns, the latter between 3.42 and 5.2 microns. The IRM employs radiative cooling, with controlled heating of critical components. The detectors are regulated to 210° ±1°K, the chopper to 240° ±0.1°K.

The signal amplifiers in all channels (40 on a preliminary engineering model) use mechanical chopping and synchronous demodulation. A multiplex switch at the output feeds an A/PW converter for digitization in the SDS.

APPENDIX

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A. LANDER ASSUMPTIONS

The lander is 3-axes stabilized and is designed to soft land on Mars after separating from the orbiter in Mars orbit. The lander system configuration and characteristics assumed for study purposes are summarized below. Figure 9A-1 shows the general arrangement, but is not to scale.

1. Lander Weight

The lander system weight at launch is 1800 lbs maximum excluding the adapter, the separation mechanisms, and adapter supported umbilical harness and possible launch environment instrumentation. Internal structure at the adapter interface is designed to apply lander loads, about all axes, to the eight vertices of the truss.

2. Lander Mass Properties

The lander center of gravity is nominally on the roll axis z, 38.4 inches from the interface separation plane in the +z direction. The maximum deviation of the c.g. from the roll axis is not specified, but will not exceed a value to be determined from tip-off constraints, discussed in Appendix B. Moments of inertia about the pitch and yaw axis are listed below, but products of inertia are unspecified.

At launch:	$I_{zz} = 418$ slug ft^2	(JPL coordinate axes)
	$I_{xx} = 275$ slug ft^2	(JPL coordinate axes)
Lander less bioshield cap:	$I_{zz} = 379$ slug ft^2	(JPL coordinate axes)
	$I_{yy} = 252$ slug ft^2	(JPL coordinate axes)

3. Lander Electrical Power Requirements

The electrical power demand of the lander is shown in Table 9A-1 for launch, cruise, and orbit phases of flight. The preseparation loads shown in the bottom row, however, are for lander internal power.

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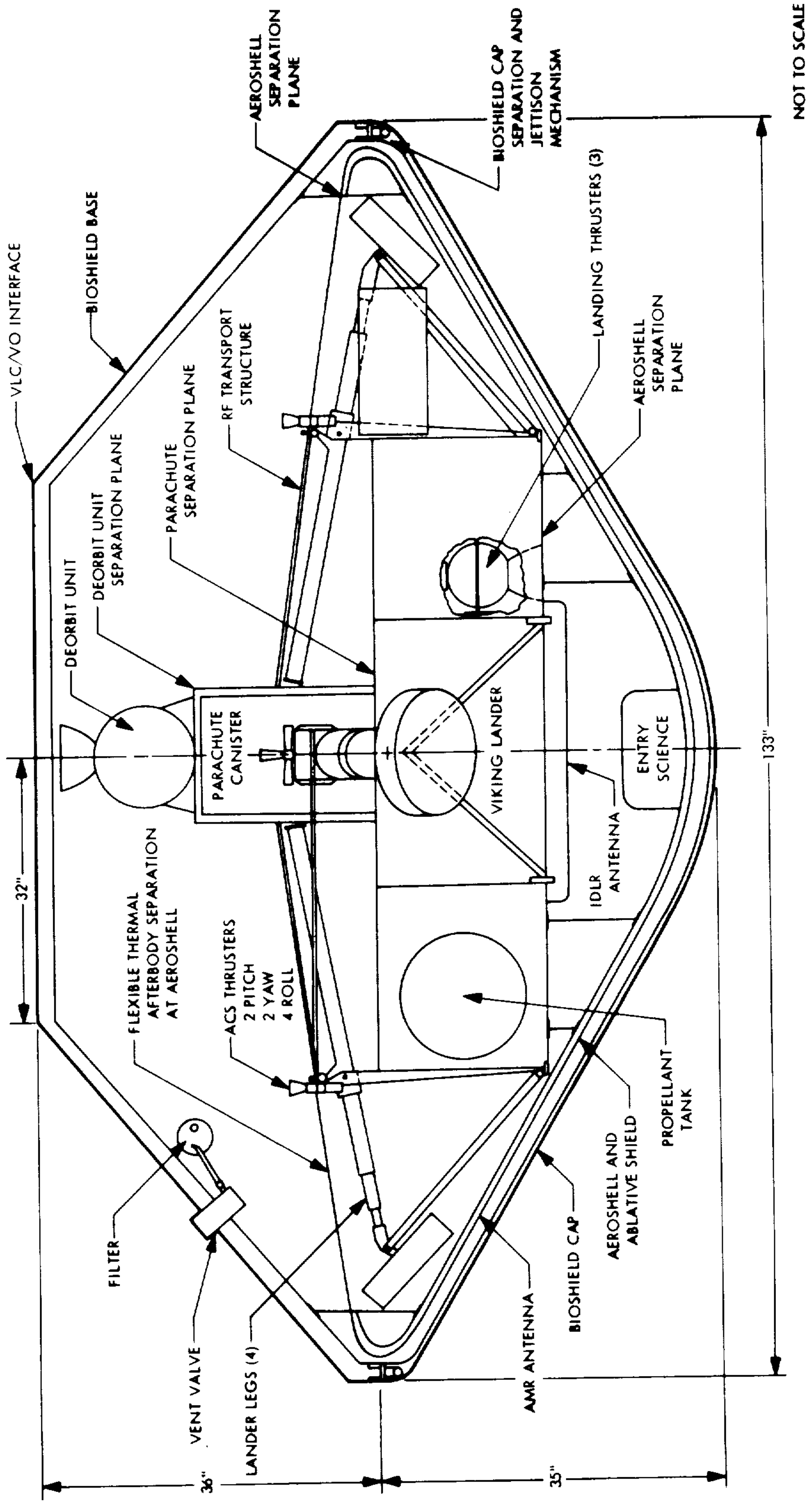

Figure 9A-1. General Arrangement of the Lander Capsule

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Table 9A-1. Electrical Power Demand of the Lander

	MISSION PHASE	POWER (watts) MIN	POWER (watts) MAX	DURATION	NOTES
1.	LAUNCH	5.75	15.75	CONTINUOUS	
2.	CRUISE				
a.	BATTERY CHARGER OFF	5.2	15.2		1
b.	BATTERY CHARGER ON	5.2	25.2		2
c.	MIDCOURSE MANEUVER	5.0	5.0	DURATION OF MANEUVER	
d.	STATUS TELEMETRY	62.7	82.7	1 hr EVERY 3 days	
3.	IN ORBIT				
a.	CRUISE BEFORE C/O	SAME AS 2a AND 2b			
b.	CHECKOUT	172	182	3 hr	3
c.	CRUISE AFTER C/O	7.5	27.5	CONTINUOUS	4
d.	PRESEPARATION	139.2	149.2		5

NOTES:

1. THERMAL CONTROL POWER 0 TO 10 watts; COMMUNICATIONS 4.5 watts; BATTERY CHARGER 0.67 watts
2. THERMAL CONTROL POWER 0 TO 10 watts; COMMUNICATIONS 4.5 watts; BATTERY CHARGER 10.7 watts
3. APPROXIMATELY 2 days BEFORE SEPARATION. ASSESSMENT AND EVALUATION REQUIRE FROM 3 TO 12 hr
4. INCLUDES 2.3 watts OF SCIENCE
5. LANDER SWITCHES TO THIS INTERNAL POWER 30 min BEFORE SEPARATION

4. Relay Link

The lander to orbiter radio relay link is VHF. During descent to the surface the radiated power is five watts. For surface operations following touchdown, the lander switches to high power such that after 90 days on the surface the radiated power is no less than 30 watts. The on-axis relay antenna gain is +5 db throughout the mission.

Following touchdown the lander relays data only when alerted by the orbiter. On data relay passes, the lander relays data for a period of not less than 12 minutes during which time the orbiter is 15 degrees or more above the lander horizon. The link has a capability of handling 10^7 bits per Martian day.

5. Telemetry

In transit, lander telemetry is transmitted over the orbiter S-band radio. Nominal bit rates of the lander telemetry system are 1050 bits per hour during cruise and 1348 bps during lander checkout.

6. Retropropulsion Maneuver

The entry angle will lie between 15 and 20 degrees (3σ). The maximum deflection velocity capability is 200 meters per second. The descent time from deorbit to touchdown is less than eight hours.

Following lander separation in orbit, the lander retro rocket will not be fired until the lander is 100 meters, or more, from the orbiter. If metallic particles can be expected in the exhaust products, this separation distance constraint will increase.

7. On-Pad Checkout

The alternate modes by which lander checkout can be accomplished on pad are by:

1) Hardlines to the spacecraft umbilical
2) VHF air link from an antenna on the umbilical tower
3) Orbiter S-band radio via an antenna on the umbilical tower.

Orbiter power-up is not required for modes 1). and 2).

8. Coordinate Axes

The lander body-fixed coordinate system, except for origin, is identical to that of the spacecraft:

"X" axis:	Pitch
"Y" axis:	Yaw
"Z" axis:	Roll

Polarities of the axes are shown in Figure 9A-2. The cone/clock angle inertial coordinate system is also the same, and is shown in Figure 9A-3.

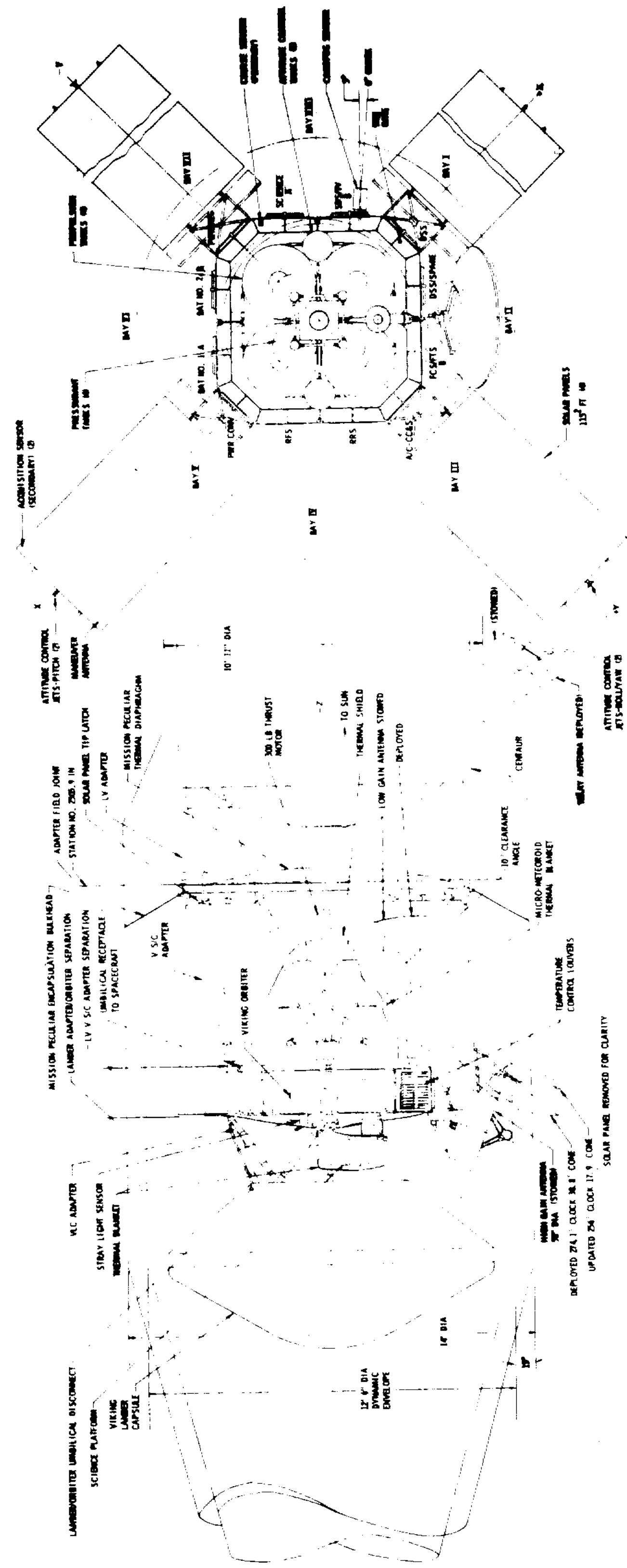

Figure 9A-2. Body-Fixed Coordinate System

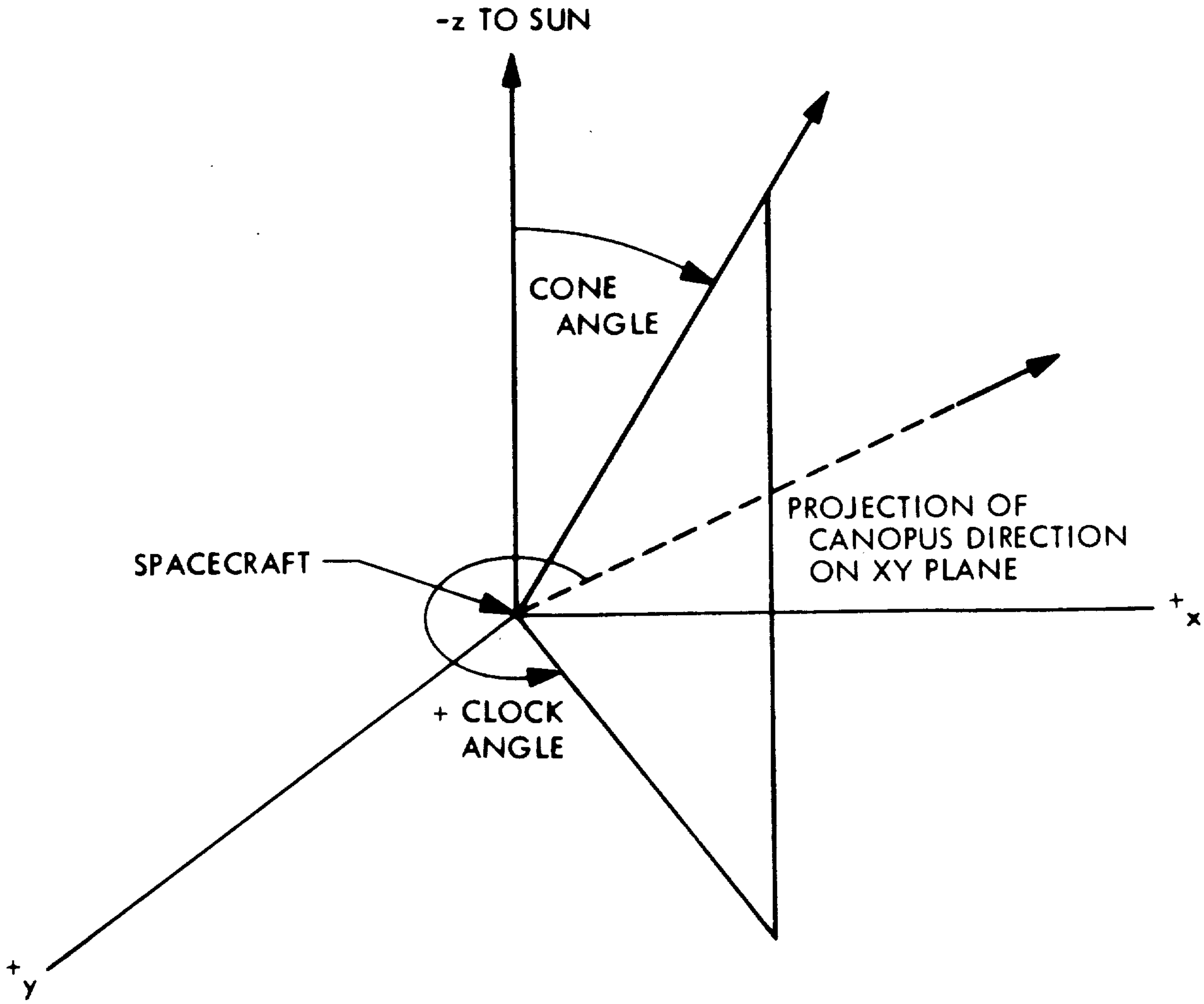

Figure 9A-3. Cone/Clock Angle Inertia Coordinate System

B. LANDER/ORBITER INTERFACE ASSUMPTIONS

1. Lander Capsule Adapter

The lander capsule is located above the orbiter in the launch configuration. In transit, the lander capsule is on the shady side of the spacecraft. The lander capsule adapter, which structurally interconnects the lander capsule and orbiter, is a truss to which the following components are attached:

1) Ejection springs and latch mechanisms (at orbiter interface)
2) Umbilical cable (electrical and RF)
3) Umbilical disconnect fitting (at orbiter interface)
4) Instrumentation for launch environment

The structural interface between the lander and lander adapter is a bolt circle field joint. A sketch of the spacecraft is shown in Figure 9B-1.

2. Separation Systems for Lander and Lander Capsule Adapter

The lander will be separated from the orbiter at a minimum relative velocity of 0.5 meters per second. Asymmetries in ejection spring and umbilical disconnect forces, and in the lander center of gravity, will be limited such that angular tip-off rates about the pitch and yaw axes will not exceed one degree per second (3σ), and about the roll axis 0.1 degree per second (3σ). Ejection of the bioshield cap will not induce angular rates about any axis in excess of 0.1 degree per second (3σ). A spring loaded umbilical disconnect fitting is assumed at the truss interface with the orbiter.

3. Disturbing Torques

At any time between launch and lander separation, disturbing torques about any spacecraft axis due to lander outgassing, momentum exchange devices, or other sources within the lander will not exceed 50 dynecentimeters.

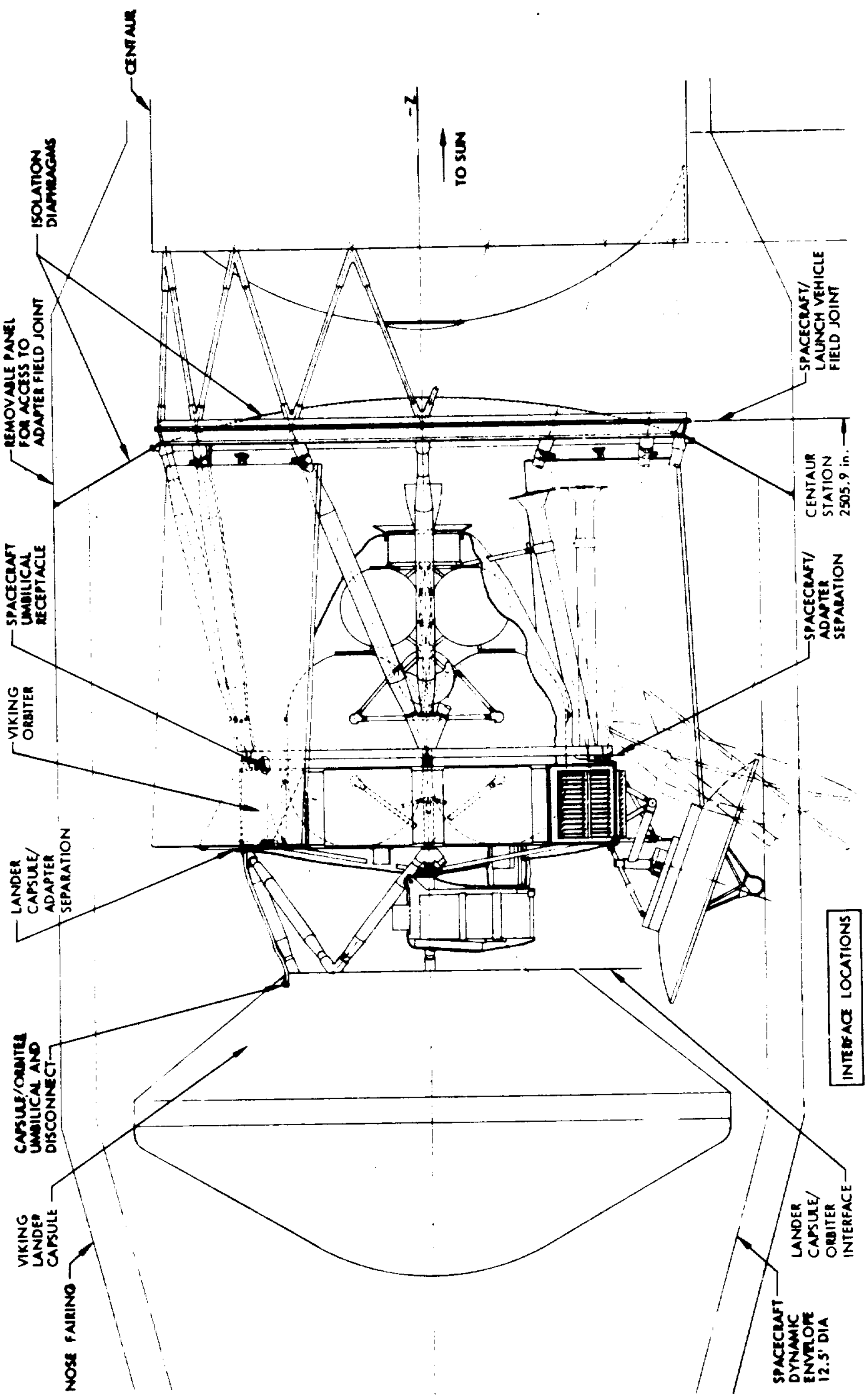

Figure 9B-1. Lander/Orbiter Interface

4. Temperature Control Interaction

Orbiter and lander capsule temperature control systems will be decoupled to the maximum feasible degree by maintaining a near-adiabatic thermal interface. For thermal control of the orbiter following release of the lander, the inside of the bioshield base has an emissivity not to exceed 0.2.

5. Radiation Environment

If radioisotope heaters are used in the lander thermal control system, or RTG's in the electrical power system, the resulting gamma and neutron fluxes and magnetic field strength will not exceed those values, to be determined, that degrade orbiter science instruments.

The lander capsule will tolerate at least 50 watts of RF power at S-band radiated into the metal nose fairing; and, conversely, the orbiter will tolerate 30 watts at VHF.

6. Lander Electrical Interface

Inflight switching of the lander from orbiter power to internal power, and conversely, will be accomplished by the orbiter central computer and sequencer (CC&S). Provision is made for short circuit protection and prevention of reverse flow of power. A ground command back-up mode will be provided. During on-pad checkout the switching from ground power to internal power does not require the orbiter to be in a power-up mode.

The lander timer and sequencer is started in flight by the orbiter prior to lander checkout and separation. The enable event is initiated by the orbiter CC&S; this CC&S event can be updated by ground command.

Power for all lander electroexplosive devices will be provided by the lander. No squib firing lines will cross the lander/orbiter interface.

The umbilical harness below the lander in-orbit disconnect fitting will be provided by the orbiter, and will contain RF and electrical cables. Some of the cables will connect to orbiter harnesses, and others will extend externally past the orbiter bus, and into the inflight umbilical disconnect fitting at the launch vehicle interface. In the later group is included the instrumentation wiring from the lander capsule adapter.

Below the launch vehicle interface, the umbilical harness splits according to whether the circuits connect with the launch vehicle telemetry or sequencer systems, or to the external umbilical connector in the side of the Centaur adapter.

7. Coordinate Axes of Lander, Orbiter, Spacecraft and Launch Vehicle

The orbiter and the lander are assumed to have, except for origin, identical body fixed coordinate systems and cone/clock angle inertial systems, as identified in Figure 9A-2, and 9A-3.

8. RF Interface

Prior to lander separation, lander telemetry is transmitted over video pairs between the lander and orbiter. A portion of this cable is incorporated in the lander capsule umbilical harness. After separation, and until the bioshield base is jettisoned, the high data rate RF link is maintained by means of a "separation antenna" which is a small cup turnstile. The antenna is used only to receive, never to transmit. After the bioshield base and adapter are jettisoned, the relay antenna is used for the remainder of the capsule descent phase and for subsequent relay passes over the lander.

The separation antenna is located on the inside surface of the bioshield base on the roll axis. The attached coaxial cable is part of the umbilical harness, and contains an attenuator. Thus only the antenna is involved in the lander sterilization process.

C. PROPULSION VELOCITY INCREMENT TRADE STUDIES

Two propulsion subsystem trades were studied for the present design weights included in section VII D and a VLC weight of 1800 lbs. The ΔV allocations are given as follows:

Maneuver	ΔV Allocation (m/sec)
Midcourse (total	15
Pre-separation orbit trims (total)	50
Post separation orbit trim (total)	50
Gravity losses (total)	110
Orbit insertion (impulsive)	1350

The first case utilized the VO weight (including contingencies), sans propulsion subsystem, of 1430 lbs. With the propulsion inerts sized for this case, the second case used the non-contingency orbiter weight of 1295 lbs.

1) Case 1: VO weight 1430 lbs (baseline weight including VLC adapter).

Propulsion subsystem weight		3980 lbs
Propellant	3130 lbs	
Inerts	850 lbs	
V S/C injected weight		7210 lbs
Orbit insertion ΔV		1350 m/sec

2) Case 2: Case 1 inerts off loaded to retain orbit insertion ΔV = 1350 m/sec with non-contingency orbiter weight (1295 lbs) including VLC adapter.

This case provides a savings of about 470 lbs in injected weight while retaining the maximum ΔV capability. However, no orbiter weight contingencies are provided.

Propulsion subsystem weight		3650 lbs
Propellant	2862 lbs	
Inerts (sans contingency)	788 lbs	
V S/C injected weight		6743 lbs
Orbit insertion ΔV		1350 m/sec

Table 9C-1. Viking ΔV Trade Studies

Parameters \ Case	1	2
VS/C Injected Weight (lb)	7210	6743
Capsule Weight (lb)	1800	1800
VLC Adapter + VO Inerts (lb) Sans Propulsion Subsystem	1430	1295
Contingencies: Propellant / Inerts	yes / yes	no / no
Orbit Insertion ΔV (m/sec)	1350	1350
VO Propulsion Inerts	850	788
VO Propulsion Propellants	3130	2862

Table 9C-1. Viking V Trade Studies

Parameters / Case	1	2
VS/C Injected Weight (lb)	7210	6743
Capsule Weight (lb)	1800	1800
VLC Adapter + VO Inerts (lb) Sans Propulsion Subsystem	1430	1295
Contingencies: Propellant / Inerts	yes / yes	no / no
Orbit Insertion ΔV (m/sec)	1350	1350
VO Propulsion Inerts	850	788
VO Propulsion Propellants	3130	2862

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